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FEASIBILITY STUDY FOR A MICROWAVE-POWERED OZONE SNIFFER AIRCRAFT

VOLUME II

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Appendix A.1

Computer Aided Design

Using three dimensional design techniques and the Advanced Surface Design software on the Computervision Designer V-X Interactive Graphics System, the aircraft configuration was created. The canard, tail, vertical tail, and main wing were created on the system using 'Wing Generator', a Computervision based program introduced in Appendix A.2. These plots can be seen in Figures A.1.1 - A.1.3. The individual components of the plane were created separately and were later individually imported to the master database. An isometric view of the final configuration can be seen in Figure A.1.4.

Figure A.1.1

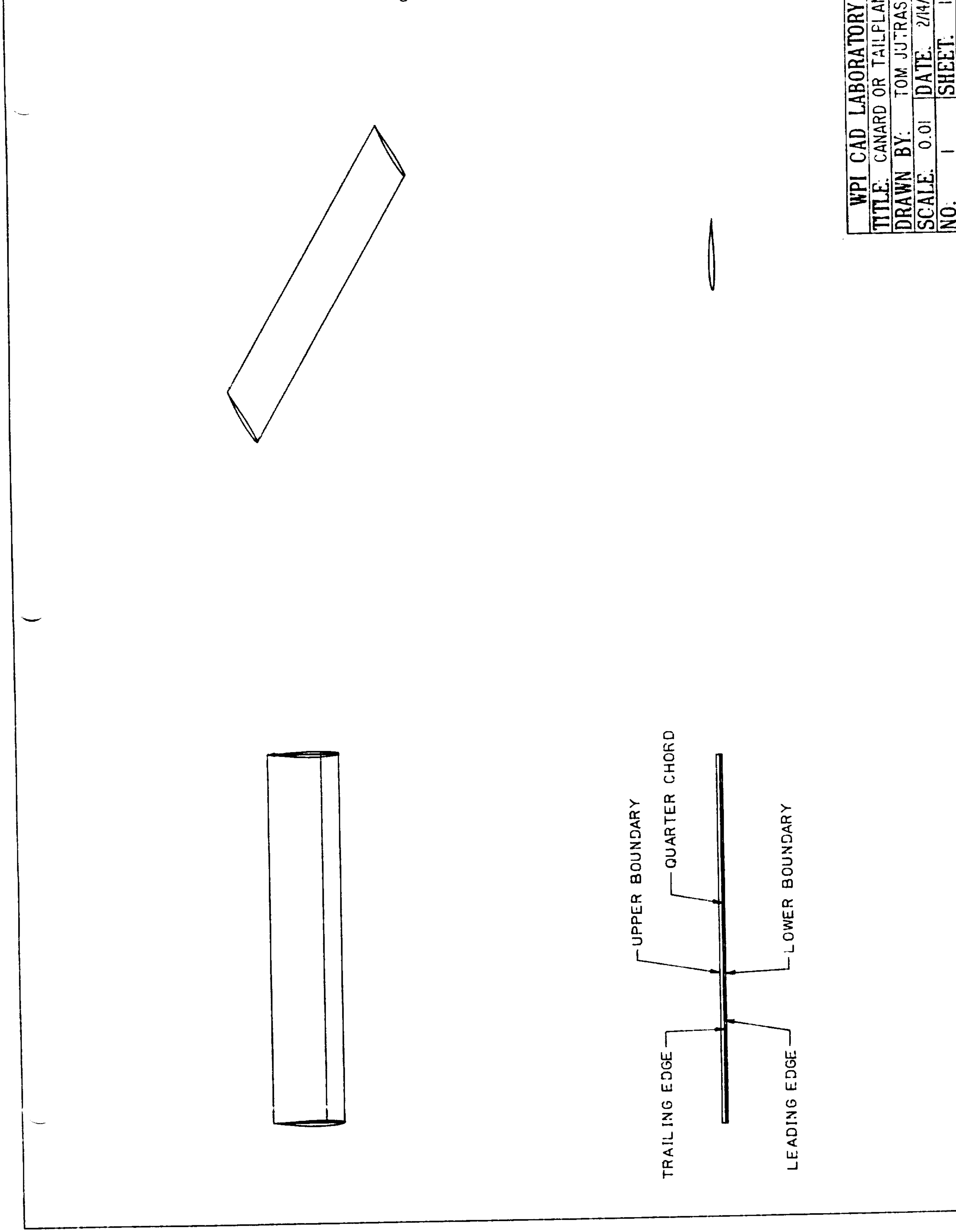
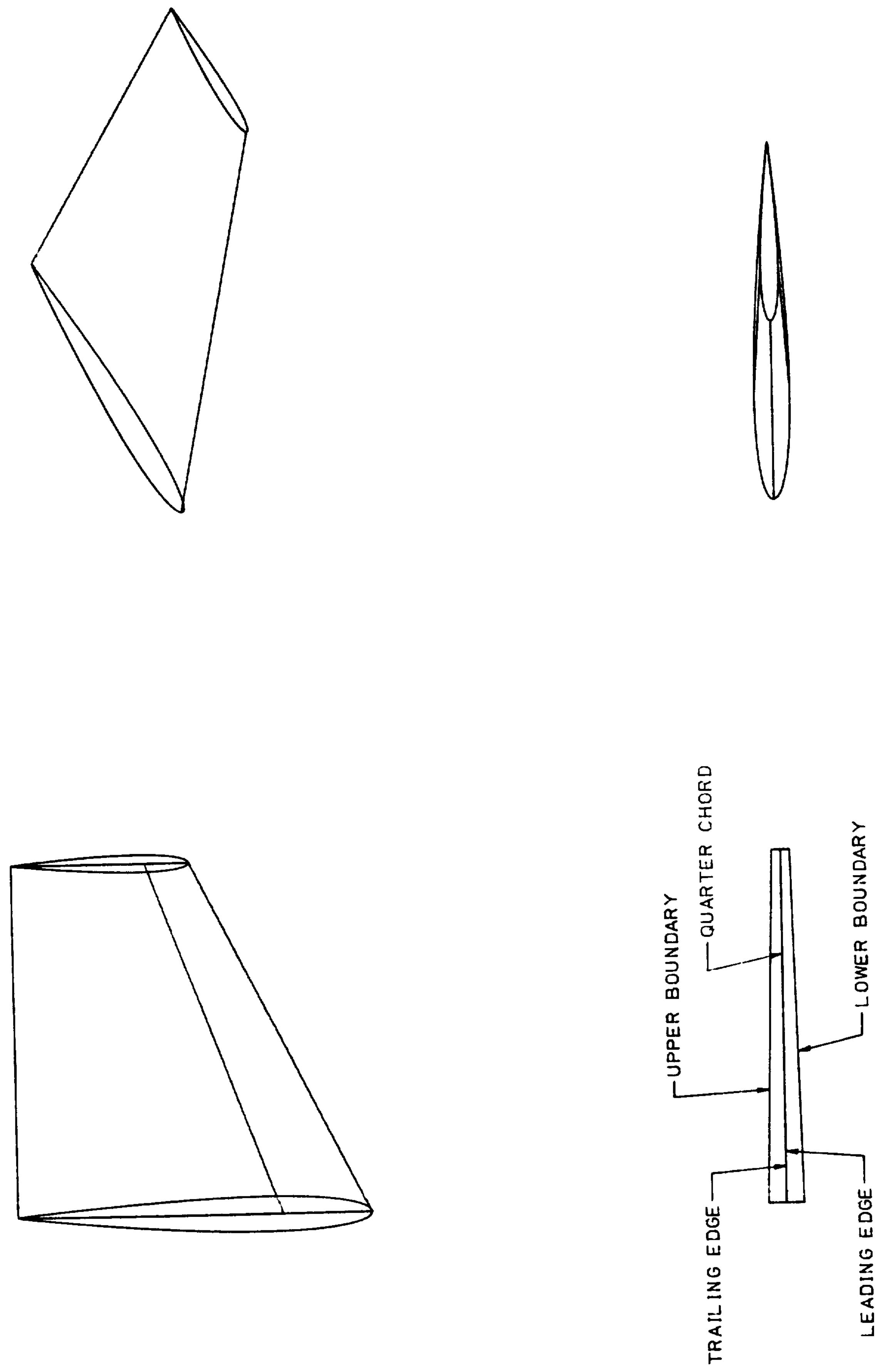

WPI CAD LABORATORY
TITLE: CANARD OR TAILPLANE
DRAWN BY: TOM JUTRAS
SCALE: 0.01
DATE: 2/14/90
NO: 1
SHEET: 1
UPPER BOUNDARY
QUARTER CHORD
LOWER BOUNDARY
TRAILING EDGE
LEADING EDGE

Figure A.1.2



WPI CAD LABORATORY	
TITLE: VERTICAL WING	
DRAWN BY: TOM JUTRAS	
SCALE: 0.02	DATE: 3/25/90
NO: 1	SHEET: 1

Figure A.1.3

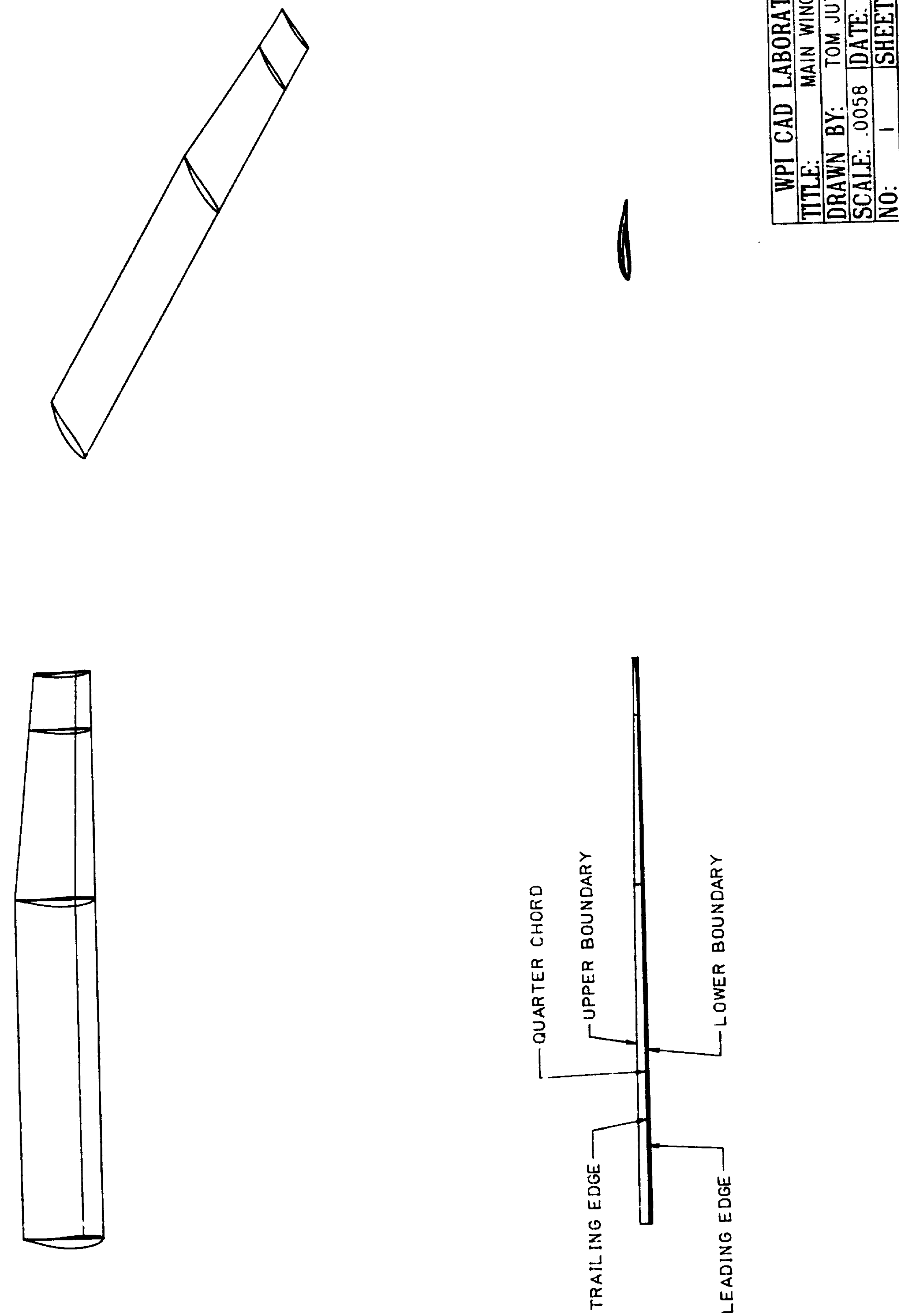
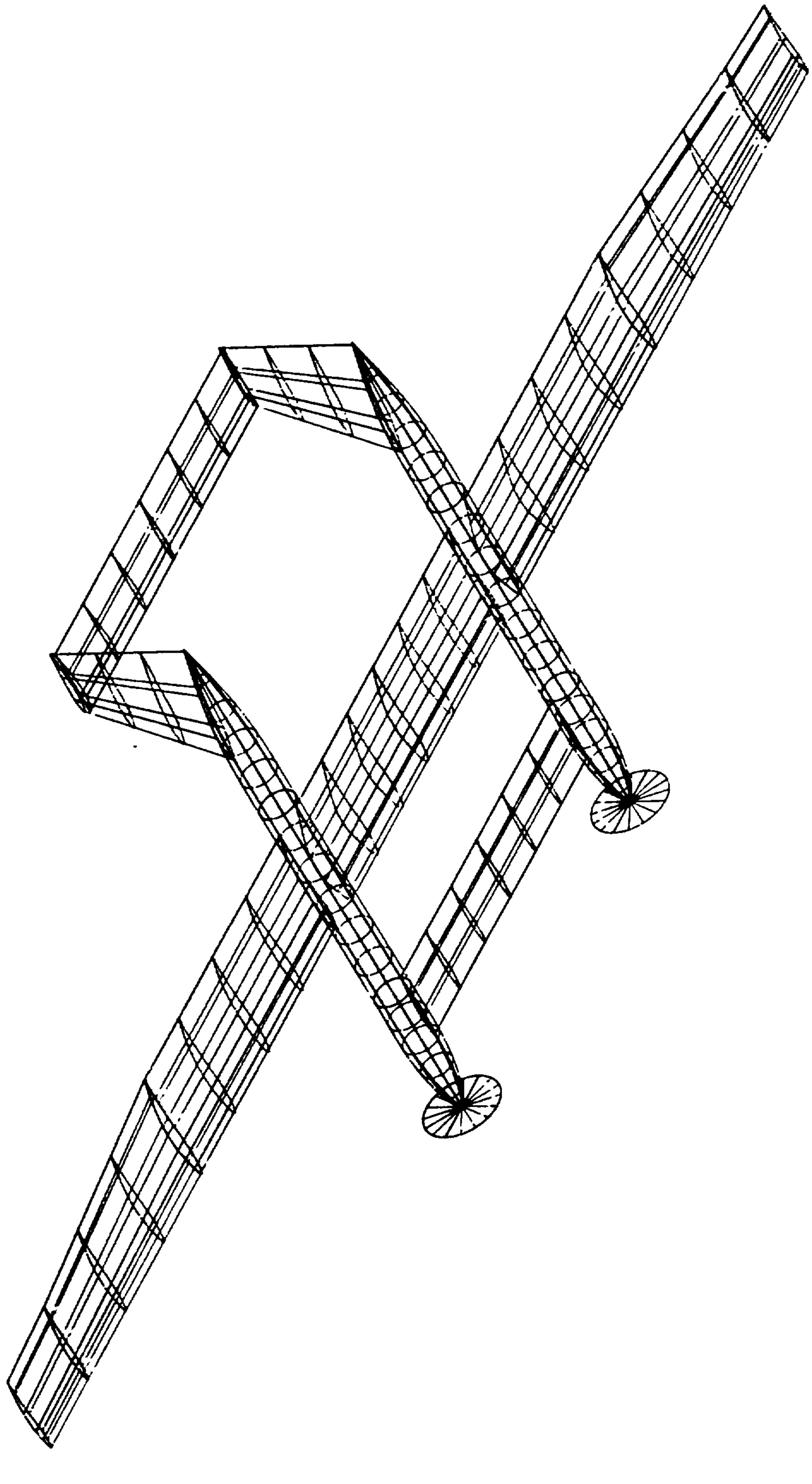

WPI CAD LABORATORY
TITLE: MAIN WING
DRAWN BY: TOM JUTRAS
SCALE: .0058
DATE: 4/2/90
NO: 1
SHEET: 1
QUARTER CHORD
UPPER BOUNDARY
LOWER BOUNDARY
TRAILING EDGE
LEADING EDGE

Figure A.1.4


WPI CAD LABORATORY
TITLE: FINAL DRAFT - ISO VIEW
NO: 1
SCALE: .004
DATE: 4/29/90
DRAWN BY: TOM JUTRAS
SHEET: 1

Appendix A.2

CAD Wing Design Program

INTRODUCTION

The creation of a complex entity such as an airplane wing can be a complex and time consuming task on a CAD system. This program allows the designer a simpler, faster, automated means by which to experiment with wing designs and create wing models. These models may then be used for creating plots, defining mass properties, and doing aerodynamic and structural analyses as necessary.

In general, a wing's airfoil or cross section is a function of the spanwise location. For this reason, an input file may be created which defines up to ten airfoils to be used along the span of the wing. If one does not wish to create airfoils (ie. a simple wing geometry), a default file containing symmetrical airfoils may be utilized instead. At spanwise locations where an airfoil section is not specified, a linear interpolation provides a smooth transition of the wing surface.

The program format requires that the user input some basic data on the dimensions of the wing; it then responds with a CAD model of the wing, and provides plots of all views of the wing.

PROGRAM THEORY

A basic design assumption for Wing Generator 1.0 is a constant taper ratio, defined by the lengths of the tip and root chord airfoils. Once these parameters are defined, the chord length of any intermediate airfoil is known; it is a function of the span location only. As airfoils in the input file are of length unity, they may simply be scaled by the taper ratio at the required span location.

After being scaled to the correct size, the coordinates of each airfoil are adjusted to reflect the angle of twist defined by the user. At this stage, the cartesian coordinates of the airfoil sections are stored in a database; they are then connected together using the CAD BSPLINE entity and copied into model space to form a complete wing. Finally the program interacts with the user to detail the wing and plot different views.

WING DESIGN ILLUSTRATION

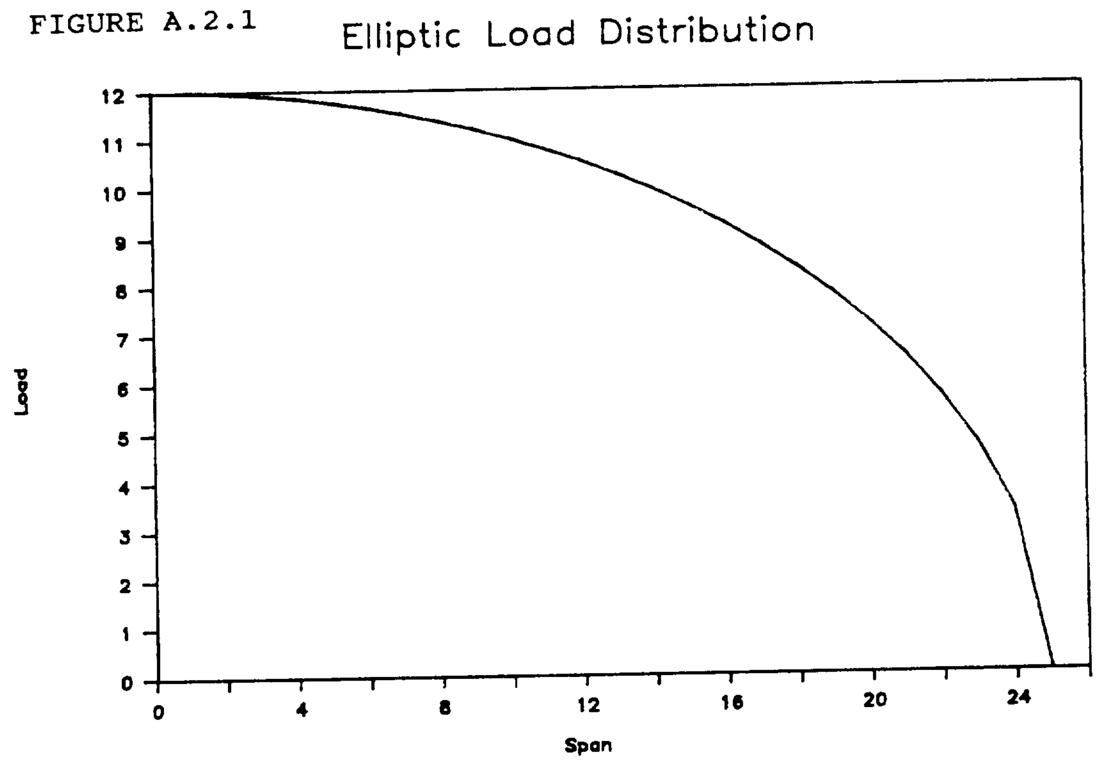
In order to demonstrate the graphic capabilities of the program, a wing was designed with an elliptic span load distribution. It was to be a forward swept wing, with the following characteristics:

Wing Span : 50 ft.
Wing Area : 750 sq. ft.

Root Chord : 20 ft.
Tip Chord : 10 ft.
Sweep Angle : -30 degrees
Lift Coeff. : 1.00
No. Airfoils: 6

Knowing the number of airfoils and total lift coefficient for the wing, an expression was derived for the local spanwise lift coefficient, (C_l). For any elliptically loaded wing, the load at any spanwise position is given by equation (A.2.1). (See Fig. A.2.1)

FIGURE A.2.1



$$W(y)=W_0\sqrt{1-(y/(b/2))^2} \tag{A.2.1}$$

However, the load at the root is given by eq. (A.2.2).

$$W_0=C_L\tfrac{1}{2}\rho V^2 S(4/\pi b) \tag{A.2.2}$$

and the total lift on the wing by (A.2.3)

$$L=\int_0^{b/2} c(y)\cdot c_1(y)dy\cdot \tfrac{1}{2}\rho V^2 \qquad \text{(A.2.3)}$$

integrating eq. (A.2.3) yields

$$L=C_L\cdot \tfrac{1}{2}\rho V^2 S \qquad \text{(A.2.4)}$$

and

$$C_L= (2/s)\int_0^{b/2} c(y)\cdot c_1(y)dy \qquad \text{(A.2.5)}$$

Substituting equation (A.2.2) into equation (A.2.1) yields

$$W(y)=C_L\cdot \tfrac{1}{2}\rho V^2 S\cdot (4/\pi b)\sqrt{1-(y/(b/2))^2} \qquad \text{(A.2.6)}$$

Substituting equation (A.2.5) into equation (A.2.6),

$$W(y)=C_1(y)\cdot c(y)\cdot \tfrac{1}{2}\rho V^2 \qquad \text{(A.2.7)}$$

Dividing both Eq. (A.2.6) and (A.2.7) by the dynamic pressure and solving for the local spanwise lift coefficient,

$$c_1(y)=C_L\cdot S\cdot (4/\pi b)\cdot \frac{\sqrt{1-(y/(b/2))^2}}{c(y)} \qquad \text{(A.2.8)}$$

Thus for the wing described above, the local lift coefficient may be given as eq. (A.2.8), provided C(y) is given by:

$$C(y)=C_r-(y/(b/2))[C_r-C_t] \tag{A.2.9}$$

where C_r is the root chord

and C_t is the tip chord.

Substituting the values for this wing into eq. (A.2.8), the following relation is found:

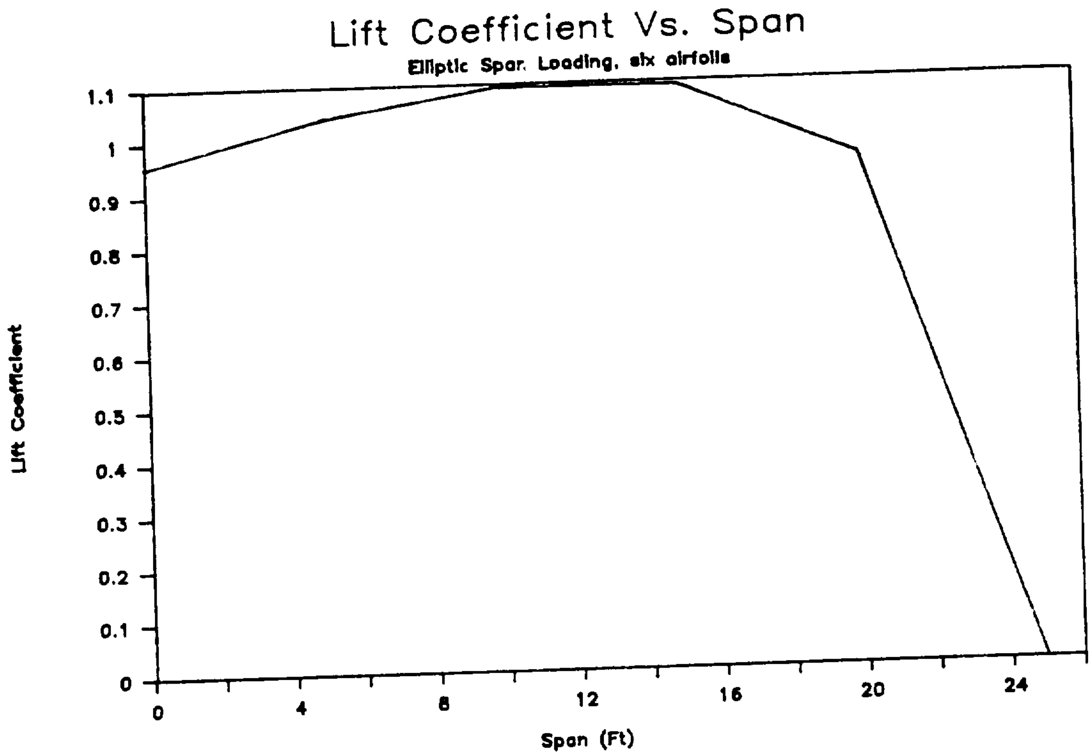
$$C_l(y)=19.10\cdot\frac{\sqrt{1-(y/25)^2}}{(20-0.4y)} \tag{A.2.10}$$

The data for the six airfoils as derived from eq. (A.2.10) are given below:

Table A.2.1

Span Location (y)	Lift Coefficient (Cl)
0	0.955
5	1.040
10	1.094
15	1.091
20	0.955
25	0.000

Figure A.2.2



Wing generator 1.0 requires actual airfoil coordinates as input. Thus an intermediate step is required, that of creating airfoil section coordinates from a target lift coefficient. The program 'Design.for' (Reference 13) was utilized for this task, the task of creating some simple airfoils of varying geometry along the wing span.

'Design.for' uses target pressure distributions, given as input, to determine airfoil section coordinates. Examples of input pressure distributions and format of output are shown at the end of this appendix.

Target pressure distributions were prescribed which would yield a value in Table (A.2.2) for that airfoil. (The output from 'Design.for' for these airfoils may be found at the end of this appendix).

Finally, the airfoil coordinates from 'Design.for' were merged and formatted as input to the Wing Generator. (The method for transferring this file from the PC to the CAD system is described in this appendix (see 'File Transfers'). The program was run for the parameters of the forward-swept wing and airfoils described previously; plots are included at the end of this appendix.

USING WING GENERATOR 1.0

Wing generator 1.0 is a fully documented computer program, written with the aircraft designer in mind. This step by step guide should aid the first time user to run the program successfully.

computer: Please enter the date.

user: Type the date.

computer: Please enter the wing span.

user: Type the length of the wing from tip to tip.

computer: Please enter the length of the root chord.

user: Type the length of the airfoil located at the fuselage.

computer: Please enter the tip chord length.

user: Type the length of the airfoil furthest from the fuselage.

computer: Please enter the sweep angle in degrees.

user: Type the sweep angle using the convention that a positive angle indicates a rearward swept wing.

computer: What is the airfoil section filename? (Default = (P.THJ.FOIL)

user: Hit the enter key to choose the default airfoil section, or type in another previosly created data file. The default file is made up of simple symmetrical airfoils. The procedure to make up

new data files will be covered in the section titled File Transfers.

<u>computer:</u> There are: X airfoils contained in the file.

Airfoil section (1) has been uploaded.

At how many span stations will (1) appear?

<u>user:</u> The computer has now read in all "X" of the airfoils. It now runs through each airfoil from root to tip and asks you how many times each of these airfoils will be used to define the wing geometry. Type the appropriate number to continue.

<u>computer:</u> Please indicate the # (1) span position for airfoil 1.

Span position =?

<u>user:</u> Type in the the location of the airfoil. The location is defined as the perpendicular distance between the airfoil under consideration and the root airfoil.

<u>computer:</u> Please enter the twist angle in degrees for same.

Twist angle =?

<u>user:</u> Type the number of degrees that the airfoil under consideration is twisted. This angle is equal to the negative of the angle of attack.

<u>NOTE:</u> This routine of defining the position and twist angle for each airfoil is repeated until all of the airfoils in the data file are defined.

computer: Do you want to see the airfoil coordinates?

user: Hit enter to default a "no" answer or type "y" to see the true space coordinates printed.

computer: Do you want to see the points plotted?

user: Hit enter to default a "no" answer or type "y" to create points at each of the coordinates along the airfoil.

computer: Do you want to surface the wing?

user: Hit enter to default a "no" answer or type "y" to create a surface on the wing geometry.

computer: Activate Part <var>

user: Type the name you wish the part to have. Hit the "ctrl" and "x" keys simultaneously.

computer: Dynamic view speed 5: view

user: Use the pen to digitize a view, then use the ICU to adjust to the location that looks best. Hit ctrl x.

computer: Insert label "Leading edge":Draw/Model entity

user: Select the modifier "near" with the pen and digitize the leading edge line and the location for the label to go. Hit ctrl x.

computer: Insert label "Trailing edge":Draw/Model entity

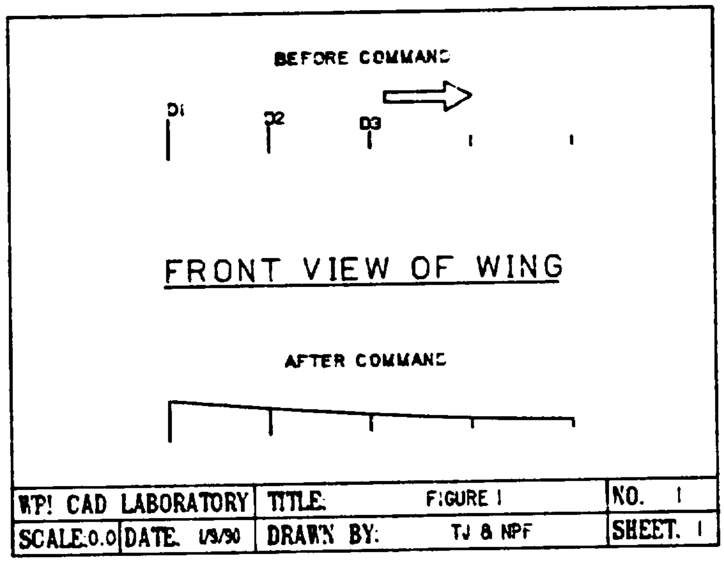
user: Repeat last step for trailing edge line.

computer: Insert label "Quarter Chord":Draw/Model entity

user: Repeat again for quarter chord line.

computer: Insert Bspline Tag = HI:Model location

user: Select the modifier "near" with the pen and then digitize the upper points on the airfoils as

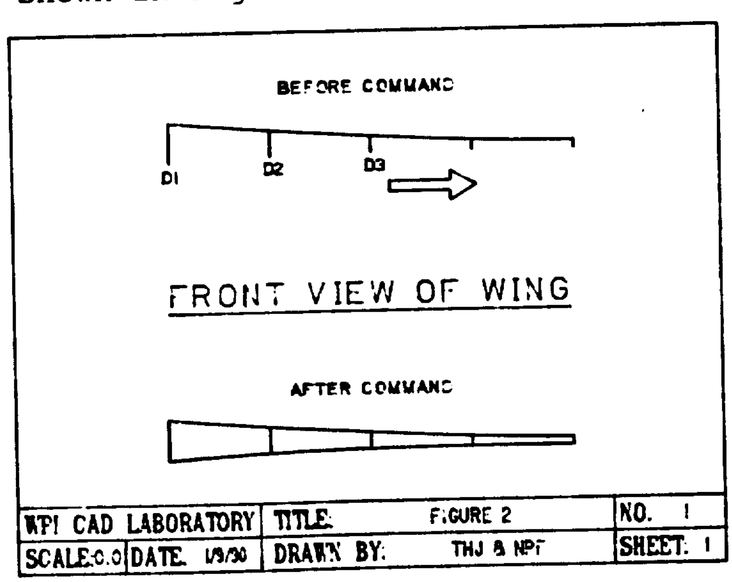


computer: Insert label "Upper Boundary":Draw/Model entity

user: Repeat previous procedure for inserting labels for the upper boundary line.

computer: Insert Bspline Tag = LOW:Model location

user: Select the modifier "near" with the pen and then digitize the lower points on the airfoils as shown in Fig. 2. Hit ctrl x.



computer: Insert Ntext ""Wing section"THJ & NPF"Scale"1 "date"1"":Draw ent

user: Use the pen to digitize any point on the title block. Hit ctrl x. This procedure will automatically configure the title block with the appropriate scale and date.

In order to have multi-color plots, entities were created on different layers in the drawing. The entities and their respective layers are as follows:

Airfoils............Layer 0

Title block.........Layer 0

Leading Edge........Layer 1

Trailing Edge.......Layer 2

Quarter Chord.......Layer 3

Upper Bound.........Layer 4

Lower Bound.........Layer 5

Rotated Airfoils....Layer 6

CONCLUSION

It has been shown that the time required to create and modify airplane wings using the Computervision Interactive Graphic System has been reduced significantly. The program Wing Generator 1.0 should prove to be an invaluable tool to the airplane designer, as it quickly and efficiently creates wing geometries which may be used as models for structural, aerodynamic, and mass properties analysis.

FORWARD SWEPT WING STATISTICS

Wing Span......................100 Ft.

Root Chord Length...............20 Ft.

Tip Chord Length................10 Ft.

Sweep Angle....................-30 Deg.

Span Station	Span Location(Ft)	Twist Angle(Deg)
1	0	-5
2	12.5	-2.5
3	25.0	0
4	37.5	2.5
5	50.0	5

INPUT FILE FORMAT

The airfoil data file may contain up to ten airfoils, each being made of fifty distinct points (25 upper, 25 lower). Each airfoil in the input file must have a chord length of unity, and twist angle of zero degrees, with the leading edge point coinciding with the origin (0,0) and the trailing edge point coinciding with (1,0). Wing Generator is formatted to accept output from the program 'Design.for' (Reference 13). The first entry in the file should be an index which represents the number of airfoils contained in the file. A sample input file is shown in this appendix.

```
Input File Format:
    AAA                  - number of airfoils in file
    XXXXXXX YYYYYYY      - coordinates of first point
    ....... .......      - rest of coordinates, 1st foil
    XXXXXXX YYYYYYY      - coordinates of last point
    XXXXXXX YYYYYYY      - 1st point, 2nd foil coordinates
    ....... .......      - etc.
```

Wing Generator Sample Input File

1	
.000000	.000000
.004278	.017986
.017037	.041356
.038060	.065242
.066987	.089596
.103323	.112707
.146447	.134173
.195619	.152904
.250000	.168457
.308658	.180038
.370590	.187144
.434737	.189127
.500000	.184654
.565263	.172402
.629410	.154952
.691342	.134813
.750000	.113580
.804381	.092567
.853553	.072772
.896677	.054881
.933013	.039328
.961940	.026240
.982963	.015517
.995722	.006885
1.000000	.000000
.995722	-.005329
.982963	-.009038
.961940	-.011078
.933013	-.011761
.896677	-.011726
.853553	-.011856
.804381	-.012942
.750000	-.015570
.691342	-.019980
.629410	-.026221
.565263	-.033723
.500000	-.041094
.434737	-.046654
.370590	-.050104
.308658	-.051724
.250000	-.051922
.195619	-.050678
.146447	-.048149
.103323	-.044114
.066987	-.038737
.038060	-.031603
.017037	-.022988
.004278	-.012153
.000000	.000000

FILE TRANSFERS

Generally data files defining the airfoils will be generated outside the Computervision system. The program being used in this project is 'Design.for' (Reference 13). In these cases, the output file of the source program must be in the form acceptable to Wing Generator 1.0, described previously. Once the file is in the correct format and saved on a floppy disk, it is ready to be transferred. The following procedure should aid the user in completing such transfers successfully.

NOTE: The user should type what is UNDERLINED.

1. On the patch panel in the CGP room (back room of the lab), connect a jumper cable between "CV9600 BAUDPCU" and "IBM XT#1" (see proctor for help).
2. Boot up the IBM XT. Type the correct date and time. Do not put a floppy diskette in the drive when booting up. Make sure the "T-switch" the IBM is on "Patch panel". At the main menu, choose the option to get to DOS level.
3. The program Kermit is used to transfer (ASCII) data from the IBM XT to the Computervision CADDS 4X system or vice versa. Enter the

Kermit program by typing

C:\>KERMIT <CR>

IBM-PC Kermit-MS V2.26

Type ? for help

Kermit-MS> (This is IBM Kermit Prompt)

4. To set the necessary default parameters, type

Kermit-MS>TAKE SET.CAD <CR>

Kermit-MS>SET BAUD 9600 <CR>

5. Place the floppy disk you want the data file transferred from in drive (A). Specify that the file be transferred from the floppy disk drive by typing

Kermit-MS>SET DEFAULT A: <CR>

6. To connect the IBM to the CV system, type

Kermit-MS>CONNECT <CR>

7. On the CV system, type

1>KERMIT <CR>

Kermit server(This is CV Kermit Prompt)

8. At this time press the CTRL and L keys simultaneously to log onto the CV system. Type your initials at the prompt

TYPE NAME, NUMBER

XXX <CR> (Your initials)

** TASK # INITIATED **

9. At this time return back to Kermit on the IBM by pressing the CTRL and] key simultaneosly and then the C key (no carriage return). The

IBM Kermit prompt "Kermit-MS>" will appear.

10. The command to begin the transfer procedure is

    Kermit-MS>SEND XXXXXXXX.XXX <CR>

11. Once the file has been transferred, type

    Kermit-MS>CONNECT <CR>

12. This will return the user to the CV Kermit prompt "Kermit server". To exit Kermit on the CV system, press the CTRL key and then the C key twice:

    Kermit server CTRL CC

13. Once at the CV system level prompt, log out by typing

    1>LOG <CR>

14. Once again return to the IBM Kermit prompt by typing

    CTRL]C (no carriage return)

15. Exit Kermit on the IBM by typing

    Kermit-MS>EXIT <CR>

    C:\>

    At this point transfer is complete.

The original filename (first level) should not be more than eight charaters in length and the catalog level (second level) not more than three characters. This file must now be renamed on the CV system to a multilevel structure so that it may be manipulated by daily save procedures. To rename the file to this

multilevel structure, the COPYTEXT command may be used at system level as follows:

1>COPYTEXT XXXXXXXX.XXX C.III.XXXXXXXX.XXX

Where "III" represents the user's initials.

Once these steps are completed, the user can run Wing Generator. When the program asks for the data filename, enter the new CV system name.

Sample Input File to 'Design.for' Program

```
NT=       25
NCP=      22
01             0.00      -1.000    +0.000
02             0.010     -1.000    +0.000
03             0.025     -1.000    +0.000
04             0.050     -1.000    +0.000
05             0.075     -1.000    +0.000
06             0.10      -1.000    +0.000
07             0.40      -1.000    +0.000
08             0.45      -1.000    +0.012
09             0.50      -0.960    +0.050
10             0.55      -0.765    +0.110
11             0.60      -0.567    +0.183
12             0.65      -0.408    +0.238
13             0.70      -0.290    +0.270
14             0.75      -0.205    +0.293
15             0.80      -0.137    +0.300
16             0.85      -0.085    +0.285
17             0.90      -0.043    +0.235
18             0.925     -0.027    +0.190
19             0.950     -0.015    +0.142
20             0.975     -0.005    +0.068
21             0.990     -0.001    +0.020
22             1.000     +0.000    +0.000 Sample
```

Output from 'Design.for' Program

```
DESIGN.FOR..M.S.GARELICK..06-09-89
MILNE-THOMSON THIN AIRFOIL DESIGN PROCEDURE
THICKNESS..CAMBER..COORDINATES
NT=25 NCP=22
INPUT CP DISTRIBUTION
XCP=    .500000 CPU= -1.193000 CPL=    .000000
XCP=    .490000 CPU= -1.193000 CPL=    .000000
XCP=    .475000 CPU= -1.193000 CPL=    .000000
XCP=    .450000 CPU= -1.193000 CPL=    .000000
XCP=    .425000 CPU= -1.193000 CPL=    .000000
XCP=    .400000 CPU= -1.193000 CPL=    .000000
XCP=    .100000 CPU= -1.193000 CPL=    .000000
XCP=    .050000 CPU= -1.193000 CPL=    .012000
XCP=    .000000 CPU= -1.193000 CPL=    .050000
XCP=   -.050000 CPU= -1.073000 CPL=    .110000
XCP=   -.100000 CPU=   -.954000 CPL=    .183000
XCP=   -.150000 CPU=   -.835000 CPL=    .238000
XCP=   -.200000 CPU=   -.716000 CPL=    .270000
XCP=   -.250000 CPU=   -.596000 CPL=    .293000
XCP=   -.300000 CPU=   -.477000 CPL=    .300000
XCP=   -.350000 CPU=   -.358000 CPL=    .285000
XCP=   -.400000 CPU=   -.239000 CPL=    .235000
XCP=   -.425000 CPU=   -.179000 CPL=    .190000
XCP=   -.450000 CPU=   -.119000 CPL=    .142000
XCP=   -.475000 CPU=   -.060000 CPL=    .068000
XCP=   -.490000 CPU=   -.024000 CPL=    .020000
XCP=   -.500000 CPU=    .000000 CPL=    .000000

CL=  1.000210
I=25 XA=   .500000 PC=  -.596500 PT=  -.596500
I=24 XA=   .495722 PC=  -.596500 PT=  -.596500
I=23 XA=   .482963 PC=  -.596500 PT=  -.596500
I=22 XA=   .461940 PC=  -.596500 PT=  -.596500
I=21 XA=   .433013 PC=  -.596500 PT=  -.596500
I=20 XA=   .396677 PC=  -.596500 PT=  -.596500
I=19 XA=   .353553 PC=  -.596500 PT=  -.596500
I=18 XA=   .304381 PC=  -.596500 PT=  -.596500
I=17 XA=   .250000 PC=  -.596500 PT=  -.596500
I=16 XA=   .191342 PC=  -.596500 PT=  -.596500
I=15 XA=   .129410 PC=  -.596500 PT=  -.596500
I=14 XA=   .065263 PC=  -.600668 PT=  -.592332
I=13 XA=   .000000 PC=  -.621500 PT=  -.571500
I=12 XA=  -.065263 PC=  -.584479 PT=  -.452195
I=11 XA=  -.129410 PC=  -.549678 PT=  -.334327
I=10 XA=  -.191342 PC=  -.500533 PT=  -.236074
I= 9 XA=  -.250000 PC=  -.444500 PT=  -.151500
I= 8 XA=  -.304381 PC=  -.382630 PT=  -.083944
I= 7 XA=  -.353553 PC=  -.315495 PT=  -.034048
I= 6 XA=  -.396677 PC=  -.242616 PT=  -.004293
I= 5 XA=  -.433013 PC=  -.167193 PT=   .007423
I= 4 XA=  -.461940 PC=  -.098740 PT=   .007918
I= 3 XA=  -.482963 PC=  -.041704 PT=   .000815
```

```
I= 2 XA=  -.495722 PC=  -.009411 PT=  -.000856
I= 1 XA=  -.500000 PC=   .000000 PT=   .000000
ALFADEG=  2.033764
CAMBERLINE SLOPES AND COORDINATES
I=25 XA=   .500000 CSLOPE=  -.776462 ZC=   .000000
I=24 XA=   .495722 CSLOPE=  -.587146 ZC=   .002916
I=23 XA=   .482963 CSLOPE=  -.395264 ZC=   .009184
I=22 XA=   .461940 CSLOPE=  -.331109 ZC=   .016819
I=21 XA=   .433013 CSLOPE=  -.264196 ZC=   .025430
I=20 XA=   .396677 CSLOPE=  -.223868 ZC=   .034297
I=19 XA=   .353553 CSLOPE=  -.180330 ZC=   .043012
I=18 XA=   .304381 CSLOPE=  -.149182 ZC=   .051113
I=17 XA=   .250000 CSLOPE=  -.113915 ZC=   .058267
I=16 XA=   .191342 CSLOPE=  -.086898 ZC=   .064157
I=15 XA=   .129410 CSLOPE=  -.054005 ZC=   .068520
I=14 XA=   .065263 CSLOPE=  -.030686 ZC=   .071236
I=13 XA=   .000000 CSLOPE=   .014020 ZC=   .071780
I=12 XA=  -.065263 CSLOPE=   .060764 ZC=   .069340
I=11 XA=  -.129410 CSLOPE=   .094334 ZC=   .064365
I=10 XA=  -.191342 CSLOPE=   .130074 ZC=   .057416
I= 9 XA=  -.250000 CSLOPE=   .156723 ZC=   .049005
I= 8 XA=  -.304381 CSLOPE=   .181352 ZC=   .039812
I= 7 XA=  -.353553 CSLOPE=   .199114 ZC=   .030458
I= 6 XA=  -.396677 CSLOPE=   .212741 ZC=   .021578
I= 5 XA=  -.433013 CSLOPE=   .216285 ZC=   .013783
I= 4 XA=  -.461940 CSLOPE=   .212537 ZC=   .007581
I= 3 XA=  -.482963 CSLOPE=   .200456 ZC=   .003240
I= 2 XA=  -.495722 CSLOPE=   .185470 ZC=   .000778
I= 1 XA=  -.500000 CSLOPE=   .178115 ZC=   .000000
ALAMBDA=   .268364 TAU=   .094476
FOURIER COEFFICIENTS
I= 1 AN= .318473D+00 BN= .173889D+00
I= 2 AN=-.920674D-01 BN= .528346D-01
I= 3 AN= .390524D-01 BN=-.255276D-01
I= 4 AN=-.598885D-01 BN=-.269270D-01
I= 5 AN= .342235D-01 BN= .557217D-03
I= 6 AN=-.318984D-01 BN= .840308D-02
I= 7 AN= .208790D-01 BN=-.110097D-02
I= 8 AN=-.238717D-01 BN=-.453781D-02
I= 9 AN= .191965D-01 BN= .108491D-02
I=10 AN=-.161761D-01 BN= .344226D-02
I=11 AN= .127289D-01 BN=-.183084D-03
I=12 AN=-.128800D-01 BN=-.207360D-02
I=13 AN= .117038D-01 BN= .285786D-03
I=14 AN=-.964384D-02 BN= .176353D-02
I=15 AN= .723877D-02 BN= .802936D-04
I=16 AN=-.722481D-02 BN=-.117083D-02
I=17 AN= .701074D-02 BN= .689152D-04
I=18 AN=-.518064D-02 BN= .112414D-02
I=19 AN= .338631D-02 BN= .977638D-04
I=20 AN=-.321564D-02 BN=-.899677D-03
I=21 AN= .339589D-02 BN=-.958598D-04
I=22 AN=-.163063D-02 BN= .744816D-03
I=23 AN=-.317677D-04 BN=-.307924D-04
```

```
I=24 AN= .142627D-08 BN=-.817194D-03
I=25 AN= .317667D-04 BN=-.307941D-04
THICKNESS SLOPES AND COORDINATES
I=25 XA=    .500000 TSLOPE=   -.134182 ZT=    .000000
I=24 XA=    .495722 TSLOPE=   -.132822 ZT=    .017475
I=23 XA=    .482963 TSLOPE=   -.128734 ZT=    .034594
I=22 XA=    .461940 TSLOPE=   -.122044 ZT=    .051008
I=21 XA=    .433013 TSLOPE=   -.112648 ZT=    .066368
I=20 XA=    .396677 TSLOPE=   -.100999 ZT=    .080351
I=19 XA=    .353553 TSLOPE=   -.086625 ZT=    .092631
I=18 XA=    .304381 TSLOPE=   -.070540 ZT=    .102918
I=17 XA=    .250000 TSLOPE=   -.051456 ZT=    .110902
I=16 XA=    .191342 TSLOPE=   -.031289 ZT=    .116318
I=15 XA=    .129410 TSLOPE=   -.006589 ZT=    .118797
I=14 XA=    .065263 TSLOPE=    .019587 ZT=    .117946
I=13 XA=    .000000 TSLOPE=    .057739 ZT=    .112885
I=12 XA=   -.065263 TSLOPE=    .089432 ZT=    .103253
I=11 XA=   -.129410 TSLOPE=    .097954 ZT=    .090988
I=10 XA=   -.191342 TSLOPE=    .099756 ZT=    .078048
I= 9 XA=   -.250000 TSLOPE=    .094058 ZT=    .065363
I= 8 XA=   -.304381 TSLOPE=    .085446 ZT=    .053615
I= 7 XA=   -.353553 TSLOPE=    .074532 ZT=    .043144
I= 6 XA=   -.396677 TSLOPE=    .064434 ZT=    .034049
I= 5 XA=   -.433013 TSLOPE=    .056483 ZT=    .026135
I= 4 XA=   -.461940 TSLOPE=    .051379 ZT=    .019076
I= 3 XA=   -.482963 TSLOPE=    .048758 ZT=    .012522
I= 2 XA=   -.495722 TSLOPE=    .047660 ZT=    .006211
I= 1 XA=   -.500000 TSLOPE=    .047238 ZT=    .000000
AIRFOIL COORDINATES..XUPPER(I)=XLOWER(I)
   .000000      .000000
   .004278      .017986
   .017037      .041356
   .038060      .065242
   .066987      .089596
   .103323      .112707
   .146447      .134173
   .195619      .152904
   .250000      .168457
   .308658      .180038
   .370590      .187144
   .434737      .189127
   .500000      .184654
   .565263      .172402
   .629410      .154952
   .691342      .134813
   .750000      .113580
                .804381         .092567
   .853553      .072772
   .896677      .054881
   .933013      .039328
   .961940      .026240
   .982963      .015517
   .995722      .006885
  1.000000      .000000
```

.995722	-.005329
.982963	-.009038
.961940	-.011078
.933013	-.011761
.896677	-.011726
.853553	-.011856
.804381	-.012942
.750000	-.015570
.691342	-.019980
.629410	-.026221
.565263	-.033723
.500000	-.041094
.434737	-.046654
.370590	-.050104
.308658	-.051724
.250000	-.051922
.195619	-.050678
.146447	-.048149
.103323	-.044114
.066987	-.038737
.038060	-.031603
.017037	-.022988
.004278	-.012153
.000000	.000000

PROGRAM LISTING

```
<#
<#
<#
<#
<#
DIM X (50),Y(50),FOX (2000),FOY (2000)
<#------------------------------
<#
<#   WING GENERATOR  1.0
<#
<#   WRITTEN BY
<#            TOM JUTRAS &
<#            NOAH FORDEN
<#
<#              1989-1990
<#
<#-------------------------------
MAXCOR = 49
INC = 0
&FNAME = "P.THJ.FOIL"
ACT PART <VAR>
ACT DRA DETAIL1 FORM L.KES.C DRAW 4
SEL CPL 3
REG GRA TAG
SEL TAG ON
PRNT
PRNT PLEASE ENTER THE WING SPAN IN INCHES
PRNT
READ (WING SPAN =? ) B
PRNT
PRNT PLEASE ENTER THE LENGTH OF THE ROOT CHORD IN INCHES
PRNT
READ (ROOT CHORD LENGTH ? ) RC
PRNT
PRNT PLEASE ENTER THE TIP CHORD LENGTH IN INCHES
PRNT
READ (TIP CHORD LENGTH ? ) TC
PRNT
PRNT PLEASE ENTER THE SWEEP ANGLE IN DEGREES
PRNT
READ (SWEEP ANGLE =? )SWEEP
PRNT
XMAX=(B/2)*TAN(SWEEP)+TC
SCALE = 10/B
CHA VIE SCALE {SCALE}:NAME TOP;NAME ISO;NAME FRONT;NAME RIGHT
PRNT WHAT IS THE AIRFOIL SECTION FILENAME (DEFAULT = ({&FNAME})
READ &FNAME
OPENR 1,&FNAME
READF 1, &NUMSC
NUMSEC = &NUMSC (1,7)
PRNT
PRNT THERE ARE: {NUMSEC} AIRFOILS CONTAINED IN THE FILE.
COUNT = 1
REPEAT
```

**ORIGINAL PAGE IS
OF POOR QUALITY**

```
    NODE = 1
    REPEAT
       READF 1, &TXT
       X (NODE) = &TXT (1,10)
       Y (NODE) = &TXT (13,22)
       NODE = (NODE + 1)
     UNTIL (NODE .EQ. (MAXCOR + 1))
&ANS = "1"
PRNT AIRFOIL SECTION ({COUNT}) HAS BEEN UPLOADED.
PRNT AT HOW MANY SPAN STATIONS WILL ({COUNT}) APPEAR?
READ ({&ANS}) &ANS
ANS = &ANS (1,7)
INDEX = 1
REPEAT
<#
<# GET ALL PARAMETERS HERE
<#
PRNT
PRNT PLEASE INDICATE THE # ({INDEX}) SPAN POSITION FOR AIRFOIL {COUN
PRNT
READ (SPAN POSITION =? ) Z
PRNT
PRNT PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME
PRNT
READ (TWIST ANGLE =? )TWIST
<#
<#REST OF ROUTINE TO INSERT AIRFOILS GOES HERE
<#
XLE=Z*TAN(SWEEP)
XTE=((XMAX-RC)/(B/2))*Z+RC
CLEN=XTE-XLE
XQUAR=XLE+(.25*CLEN)
NODE = 0
REPEAT
   NODE = (NODE + 1)
   SPOT = ((100 * INC) + NODE)
   FOX (SPOT) =(X (NODE) * CLEN) +XLE
   FOY (SPOT) = Y (NODE) * CLEN
   TEMPX = FOX (SPOT)
   TEMPY = FOY (SPOT)
   ADD = 0
   IF (TEMPX .LT. XQUAR) ADD = -180
   ARG = (ADD + TWIST + ATAN (TEMPY / (TEMPX - XQUAR) ) )
   RVEC = (((TEMPX-XQUAR)*(TEMPX-XQUAR)) + (TEMPY * TEMPY)) ** 0.5
   FOX (SPOT) = (RVEC * (COS (ARG))) + XQUAR
   FOY (SPOT) = (RVEC * (SIN (ARG)))
 UNTIL (NODE .EQ. MAXCOR)
 FOX (SPOT + 1) = Z
 FOX (SPOT + 2) = TWIST
 FOX (SPOT + 3) = XQUAR
 INC = (INC + 1)
 INDEX = (INDEX + 1)
 UNTIL (INDEX .EQ. (ANS+1))
 COUNT = (COUNT + 1)
```

```
UNTIL (COUNT .EQ. (NUMSEC + 1))
<#
<#
<# THIS ROUTINE WILL PRINT THE MASTER ARRAY
<#NOTE-THE 50TH POSITION IS THE SPAN (Z) LOCATION
<#NOTE-THE 51ST POSITION IS THE AIRFOIL TWIST AT (Z)
<#NOTE-THE 52ND POSITION IS THE QUARTER CHORD POSITION AT (Z)
<#
&LST = "N"
READ (DO YOU WANT TO SEE THE AIRFOIL COORDINATES ?) &LST
IF (&LST .EQ. "N") GOTO JUMP
PRNT
PRNT
PRNT
PRNT ---AIRFOIL LIST--- TRUE SPACE COORDINATES.
PRNT
POL = Ø
REPEAT
   POL = (POL + 1)
   PRNT {POL}, {FOX(POL)}, {FOY(POL)}
UNTIL (POL .EQ. (INC * 1ØØ))
#JUMP
<#
<#
<#  ROUTINE TO PLOT BSPLINES OF AIRFOILS IN MASTER ARRAY
<#
<#
NUM  = Ø
ZNUM = 5Ø
CHECK = 48
REPEAT
INS BSPL:   <#
REPEAT
NUM = NUM + 1
X{FOX(NUM)}Y{FOY(NUM)}Z{FOX(ZNUM)},<#
UNTIL (NUM .GT. CHECK)
<CR>
NUM = NUM + 51
CHECK = CHECK + 1ØØ
ZNUM = ZNUM +1ØØ
UNTIL (ZNUM .GT. (INC*1ØØ))
DYN VIEW SPEED 5: <VAR>
&AXE = "N"
READ (DO YOU WANT TO SEE THE POINTS PLOTTED ?) &AXE
IF (&AXE .EQ."N") GOTO LEDGE
INIT FEM
TOM  = Ø
REPEAT
NOAH = 1
    REPEAT
        INS GPO1:X{FOX (NOAH + (TOM*1ØØ))}Y{FOY(NOAH + (TOM*1ØØ))}
          Z{FOX(TOM*1ØØ + 5Ø)}
       NOAH = NOAH + 5
    UNTIL (NOAH .GT. MAXCOR)
```

ORIGINAL PAGE
OF POOR QUALITY

```
   TOM = TOM + 1
UNTIL (TOM .EQ. INC)
<#
<#
<# THIS ROUTINE WILL CONNECT THE LEADING EDGES
<#
<#
#LEDGE
SEL LAY 1
NUM = 1
CHECK = INC*100
INS BSPL TAG LEDGE:  <#
REPEAT
X{FOX(NUM)}Y{FOY(NUM)}Z{FOX(NUM + 49)},<#
NUM = NUM + 100
UNTIL (NUM .GT. CHECK)
<CR>
INS LAB "LEADING EDGE":NEAR <VAR>
<#
<#
<# THIS ROUTINE WILL CONNECT THE TRAILING EDGES
<#
<#
SEL LAY 2
NUM = 25
CHECK = INC*100
INS BSPL TAG TEDGE:   <#
REPEAT
X{FOX(NUM)}Y{FOY(NUM)}Z{FOX(NUM + 25)},<#
NUM = NUM + 100
UNTIL (NUM .GT. CHECK)
<CR>
INS LAB "TRAILING EDGE":NEAR <VAR>
<#
<#
<# THIS ROUTINE WILL CONNECT THE QUARTER CHORD POINTS
<#
<#
SEL LAY 3
NUM = 52
CHECK = INC*100
INS BSPL TAG QUARCH:   <#
REPEAT
X{FOX(NUM)}Y0Z{FOX(NUM-2)},<#
NUM = NUM + 100
UNTIL (NUM .GT. CHECK)
<CR>
INS LAB "QUARTER CHORD":NEAR <VAR>
ZOOM DRA WIN:<VAR>
SEL LAY 4
INS BSPL TAG HI:NEAR <VAR>
INS LAB "UPPER BOUNDARY":NEAR <VAR>
SEL LAY 5
INS BSPL TAG LOW:NEAR <VAR>
```

```
INS LAB "LOWER BOUNDARY":NEAR <VAR>
ZOOM DRA ALL
<#
<#
<# THIS ROUTINE WILL DETAIL THE DRAWINGS
<#
<#
ERASE ENT :VIE NAME TOP TAG HI LOW
ERASE ENT :VIE NAME ISO TAG HI LOW QUARCH
ERASE ENT :VIE NAME RIGHT TAG  QUARCH
ECHO TAG OFF
SEL MOD DRA
READ (PLEASE ENTER THE DATE ?)&DATE
INS NTEXT ""WING SECTION"THJ & NPF"{SCALE}"1"{&DATE}"1"":<VAR>
ECHO APP SYM OFF
SEL MOD MOD
SEL LAY 6
NUM = 52
CHECK = INC * 100
INS POI: <#
REPEAT
   X {FOX(NUM)} Y0 Z{FOX(NUM-2)}, <#
   NUM = NUM + 100
UNTIL (NUM .GT. CHECK)
<CR>
ZOO DRA WIN: <VAR>
ROT ENT COPY AX270:<VAR>
ZOO DRA ALL
ECH LAY 6
ERASE ENT: PWIN <VAR>
ECH LAY ALL
&SURF = "N"
READ (DO YOU WANT TO SURFACE THE WING ?) &SURF
IF (&SURF .EQ. "N") END
<#
<#
<# ROUTINE TO SURFACE WING BEGINS HERE
<#
<#
CONVERT ENT :BSPL LAY 0 VWIN NAME RIGHT
SEL LAY 10
NUM = 50
POLE = 1
CHECK = INC*100
TOTAL = 49
#SURF
REPEAT
   GEN SPOLE: X{FOX(POLE)}Y{FOY(POLE)}Z{FOX(NUM)}, <#
   X{FOX(POLE+100)}Y{FOY(POLE+100)}Z{FOX(NUM+100)}
<CR>
   POLE = POLE + 1
UNTIL (POLE .EQ. TOTAL)
POLE = POLE + 52
NUM = NUM + 100
```

```
TOTAL = TOTAL +1ØØ
IF (NUM+1ØØ .LT. CHECK) GOTO SURF
```

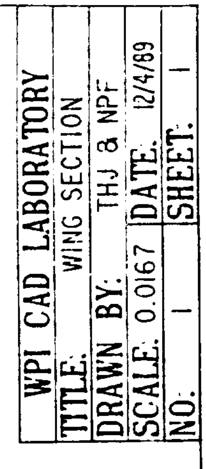
WING GENERATOR - SAMPLE RUN

```
<#
<#
<#
<#
<#
P.THJ.H.MQP1 IS THE HARDFILE

#Ø3#RUN NEW P.NPF.WIND


PLEASE ENTER THE WING SPAN IN INCHES

WING SPAN =? 12ØØ


PLEASE ENTER THE LENGTH OF THE ROOT CHORD IN INCHES

ROOT CHORD LENGTH ? 24Ø


PLEASE ENTER THE TIP CHORD LENGTH IN INCHES

TIP CHORD LENGTH ? 12Ø


PLEASE ENTER THE SWEEP ANGLE IN DEGREES

SWEEP ANGLE =? -3Ø


WHAT IS THE AIRFOIL SECTION FILENAME (DEFAULT = (P.THJ.FOIL)
P.THJ.FSWEPT

THERE ARE: 5 AIRFOILS CONTAINED IN THE FILE.
AIRFOIL SECTION (1) HAS BEEN UPLOADED.
AT HOW MANY SPAN STATIONS WILL (1) APPEAR?
{&ANS}1

PLEASE INDICATE THE # (1) SPAN POSITION FOR AIRFOIL 1

SPAN POSITION =? Ø


PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME

TWIST ANGLE =? -5

AIRFOIL SECTION (2) HAS BEEN UPLOADED.
AT HOW MANY SPAN STATIONS WILL (2) APPEAR?
{&ANS}1

PLEASE INDICATE THE # (1) SPAN POSITION FOR AIRFOIL 2

SPAN POSITION =? 15Ø
```

```
PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME

TWIST ANGLE =? -2.5

AIRFOIL SECTION (3) HAS BEEN UPLOADED.
AT HOW MANY SPAN STATIONS WILL (3) APPEAR?
{&ANS}1

PLEASE INDICATE THE # (1) SPAN POSITION FOR AIRFOIL 3

SPAN POSITION =? 300


PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME

TWIST ANGLE =? 0

AIRFOIL SECTION (4) HAS BEEN UPLOADED.
AT HOW MANY SPAN STATIONS WILL (4) APPEAR?
{&ANS}1

PLEASE INDICATE THE # (1) SPAN POSITION FOR AIRFOIL 4

SPAN POSITION =? 450


PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME

TWIST ANGLE =? 2.5

AIRFOIL SECTION (5) HAS BEEN UPLOADED.
AT HOW MANY SPAN STATIONS WILL (5) APPEAR?
{&ANS}1

PLEASE INDICATE THE # (1) SPAN POSITION FOR AIRFOIL 5

SPAN POSITION =? 600


PLEASE ENTER THE TWIST ANGLE IN DEGREES FOR SAME

TWIST ANGLE =? 5

DO YOU WANT TO SEE THE AIRFOIL COORDINATES (N)

DO YOU WANT TO SEE THE POINTS PLOTTED (N)

PLEASE ENTER THE DATE ()
                          1/10/90
DO YOU WANT TO SURFACE THE WING (N)

#03#<VAR>DYNamic VIEw SPEED 5: view d
```

```
#Ø3#<VAR>
#Ø3#<VAR>DYNamic VIEw SPEED 5: view d
#Ø3#<VAR>
#Ø3#
#Ø3#SEL LAY 1
#Ø3#INS LAB "UPPER BOUNDARY": DRAW/MODEL ent NEar  <VAR>d DRAW loc d
#Ø3#<VAR>
#Ø3#SEL LAY 5
#Ø3#INS BSP1 TAG = LOW: MODEL loc NEar  <VAR>ddddd
#Ø3#INS LAB "LOWER BOUNDARY": DRAW/MODEL ent NEar  <VAR>d DRAW loc d
#Ø3#ZOOm DRA ALL

#Ø3#ERAse ENT : MODEL ent VIe NAME TOP TAs HI LOW
#Ø3#ERAse ENT : MODEL ent VIe NAME ISO TAs HI LOW QUARCH
#Ø3#ERAse ENT : MODEL ent VIe NAME RIGHT TAs  QUARCH
#Ø3#ECHo TAG OFF

#Ø3#SEL MOD Dra

 Selected mode is DRAW.
#Ø3#ECHo APP SYm OFf
#Ø3#SEL MOD Mod

 Selected mode is MODEL.
#Ø3#SEL LAY 6
#Ø3#
#Ø3#ZOO DRA WIN: DRAW loc   <VAR>=\  DRAW loc  DRAW loc dd DRAW loc
#Ø3#ROT ENT COPY AX27Ø: MODEL ent  <VAR>d
 MODEL loc  O  POi d: MODEL loc ;#Ø3#<VAR>=####ROT ENT COPY AX27Ø:
#####Ø3#<VAR>ROT ENT COPY AX27Ø: MODEL ent d
 MODEL loc  O  POi d: MODEL ent d
 MODEL loc  O  POi d: MODEL ent d
 MODEL loc  O  POi d: MODEL ent d
 MODEL loc  O  POi d
#Ø3#<VAR>
#Ø3#ZOO DRA ALL

#Ø3#ECH LAY 6

#Ø3#ERAse ENT: MODEL ent  PWin DRAW loc   <VAR>dddddd
#Ø3#ECH LAY ALL

#Ø3#DO HARd
```

Figure A.2.3


WPI CAD LABORATORY
TITLE: WING SECTION
DRAWN BY: THJ & NPF
SCALE: 0.0167
DATE: 12/4/89
NO: 1
SHEET: 1

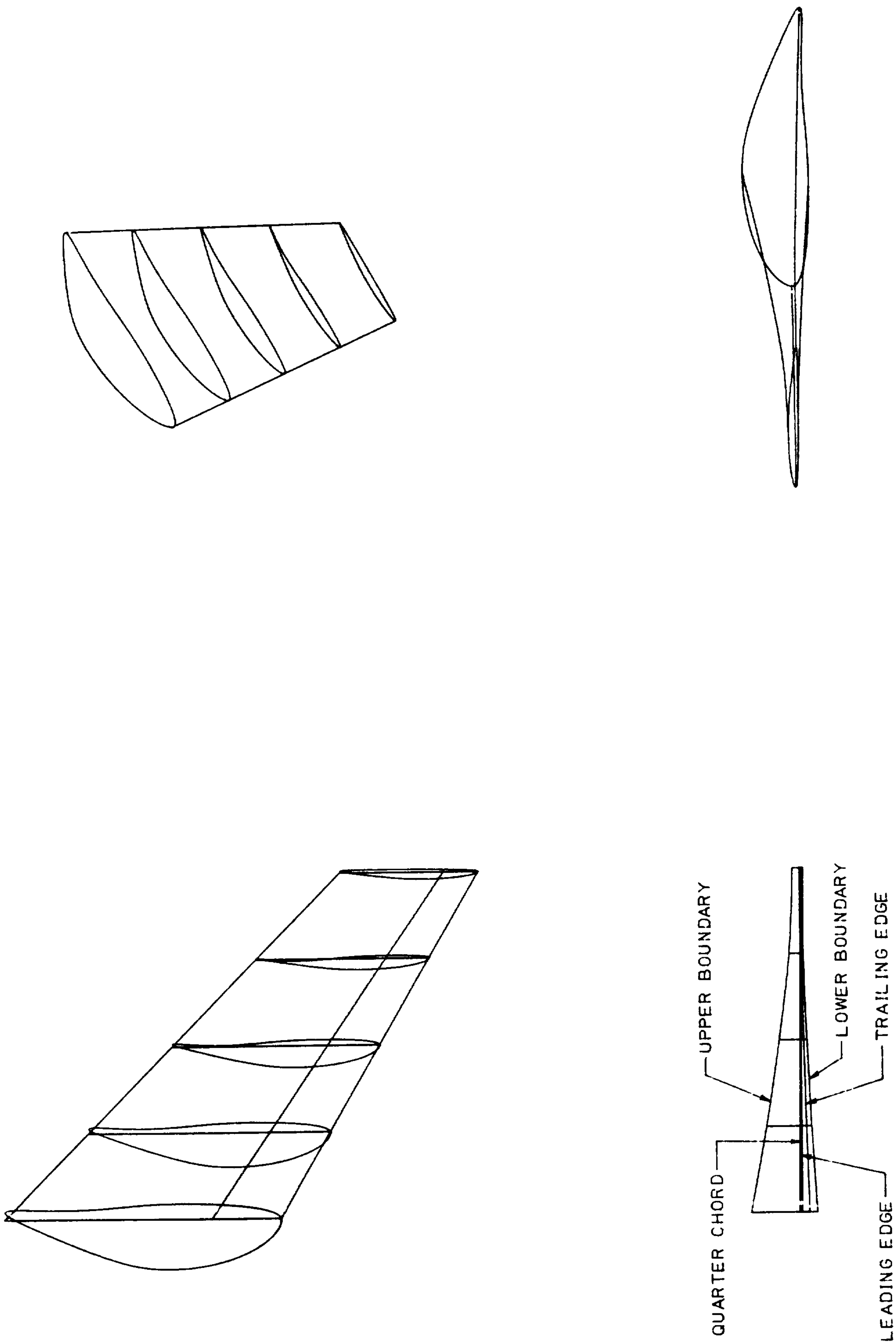
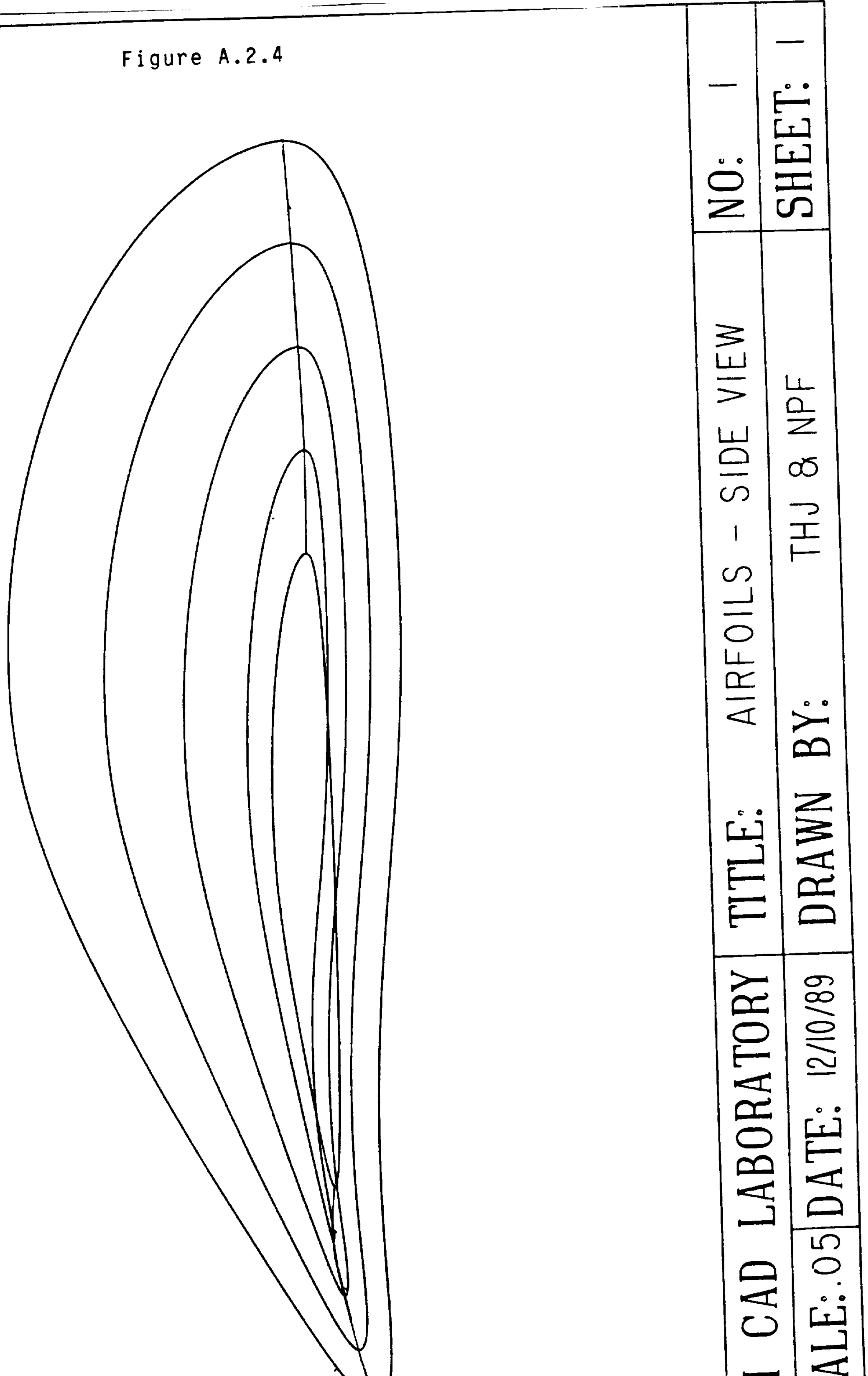

UPPER BOUNDARY
LOWER BOUNDARY
TRAILING EDGE
QUARTER CHORD
LEADING EDGE

Figure A.2.4


WPI CAD LABORATORY
TITLE: AIRFOILS - SIDE VIEW
NO: 1
SCALE: .05
DATE: 12/10/89
DRAWN BY: THJ & NPF
SHEET: 1

Appendix A.3

Weight Estimation Computer Code

The computer code asks the user to input several pieces of data before the weight iteration begins.

1.	Lift coefficient:	The value of C_L asked for here is the level flight C_L at design altitude.
2.	Mach Number Range:	This input is the values of the lower and upper mach number range as well as the step interval.
3.	Payload weight:	The weight of payload to be carried in pounds.
4.	Thickness ratio:	The maximum thickness to chord ratio entered as a decimal value.
5.	Sweep angle:	The wing quarter chord sweep in radians.
6.	Limit load factor:	The value where yielding will occur.
7.	Aspect ratio:	Value for main wing. Wing weight estimates may be low for high aspect ratios.
8.	No. of propellers:	Total number of propellers.
9.	Number of blades:	Number of blades for each propeller.
10.	Propeller diameter:	Maximum propeller diameter in feet.

After this data is entered, the program sets values for constants at the design altitude. These initial values are given in Figure A.3.1. The program will then iterate the component weights over the range of Mach numbers specified. A flowchart of the iteration method is given in Figure A.3.2.

The iteration method is based on a graph of calculated gross weight (WG) versus an initial weight

guess (WI) as shown in Figure A.3.3. The 45° line starting at the origin is the line where the initial guess equals the iterated weight. At low initial weights, the calculated weight will be greater. As the initial guesses increase, the calculated weight decreases and the line traced out will cross the line where the initial and calculated weights are equal. This is the point where the weights have converged. The program will increment initial weights (DW) until this crossover occurs. At this point the initial weight will be reset to the last value (LW) where the calculated weight is greater than the initial weight, the increment will be reduced and the procedure will begin again until the calculated value is within five pounds of the initial value.

The iterations begin with an initial weight guess of 1000 pounds, which is the payload weight. The lift to drag ratio for the initial sizing is calculated first. This calculation is accomplished as follows. The total drag is the sum of pressure and skin friction drags on the wing, tail and fuselage plus the induced drag on the wing.

$$C_D = C_{D0} + K' C_L^2 + K'' C_L^2 \qquad \text{(A.3.1)}$$

where:

$K' = 1 / (\pi * e * AR)$
K'' = Average value of NACA 2412 obtained from Reference 1

The skin friction drags, C_f, on the wing, tail and fuselage are determined based on flat plate, turbulent flow, and are referenced to the wing area by the following formula:

$$C_f = [.074 / N_{RE}^{.2}]\ S_{wet}/S_{ref} \qquad \text{(A.3.2)}$$

The total skin friction drag is then multiplied by 1.25 to account for pressure drag and mutual interference as suggested on p. 2-14 of Reference 24. The sum of the skin friction drags is the drag coefficient at zero lift, C_{DO}. The induced drag is determined using a span efficiency factor, e, of .85, and viscous drag is calculated based on an estimate from p. 479 of Reference 1. All three are summed to obtain the total drag coefficient, C_D.

The power required is calculated based on the current weight estimate. This value is simply the velocity times the drag force. The power required is multiplied by a factor of 1.22 to account for equipment efficiency, motor (97%), DC conversion (98%), propeller (85%) and 10% extra power used for climbing.

$$PR = 1.22(WI/LD)(V) \qquad \text{(A.3.3)}$$

All of the component weights are now evaluated and the size of the external rectenna, if required, is

determined. The rectenna sizing is determined by the following methodology. The total rectenna area required to absorb the necessary power is calculated as:

$$AN = PR\ /\ .0872 \qquad (A.3.4)$$

where:

AN - Total Rectenna Area (ft^2)
PR - Power Required (horsepower)
.0872 - Power Density (hp/ft^2) from Reference 6

If this area is less than or equal to the wing area then no external rectenna is needed. The weight of the rectenna on the bottom of the wing surface is estimated as .076 pounds per square foot also suggested by W. Brown in Reference 6. This weight includes the weight of the thin film covering and the reflecting plane of the rectenna. If the required rectenna area is larger than the wing area, the difference is the external rectenna area. The external rectenna is assumed to be a flat disk. The diameter is determined and the power required is re-calculated to include the drag of the external rectenna.

When convergence is reached, the program will output data to the monitor, printer and the specified data file. The output to the monitor and printer are broken down into component weights, span, lift-to-drag ratio, power available, wing area, wing loading and ultimate load factor. The output to the data file is in

ASCII format. The data file is opened as an append file so that each time the same filename is used the new data will be appended to the original file. This output includes the Mach number, lift coefficient, gross weight, wing area, external rectenna area, lift to drag ratio, optimum lift-to-drag ratio, span and power available.

```
10 REM      Weight Estimation Program for HALE Aircraft
20 REM      Written By: Scott B. Sandler
30 REM      Date: Revised November 18, 1989 V2.1
40 REM
50 REM      List of Variables and Constants
60 REM      CL - Coefficient of Lift            NRE - Reynolds Number
70 REM      M  - Mach Number                      S - Wing Area (ft2)
80 REM      LD - Lift to Drag Ratio              MU - Viscosity
90 REM      AR - Aspect Ratio                     B - Span (ft)
100 REM     TC - Thickness Ratio               CBAR - Mean Cord (ft)
110 REM     SW - Sweep Angle (rad)                N - Load Factor
120 REM     WP - Payload Weight (lbs)            CO - Drag at Zero Lift
130 REM     CD - Drag Coefficient                NP - Number of Propellers
140 REM     WA - Wgt/Thin Film on Wing (lbs)     AN - Total Rectenna Area (ft2)
150 REM     AX - Area External Rectenna (ft2)    WX - Wgt External Rectenna (lbs)
160 REM     DR - Diameter/External Rectenna      CR - Skin Friction Ext. Rectenna
170 REM     CW - Skin Friction on Wing           CU - Skin Friction of Fuselage
180 REM     F$ - Output Filename                OPT - Optimum L/D ratio
190 REM     SV - Vertical Tail Area              SH - Horizontal Tail Area
200 REM     BV - Vertical Tail Span              BH - Horizontal Tail Span
210 REM     CH - Skin Friction on HT             CV - Skin Friction on VT
220 REM      F - Iteration Flag
230 REM
240 REM *** define constants ***
250 REM
260 CLS
270 RO = 3.211E-05              :REM denisty at altitude  (lb/ft3)
280 MU = 3.15E-07               :REM Viscosity
290 SS = 1003.2                 :REM local sonic speed (ft/s)
300 E = .85                     :REM Efficiency Factor
310 GW = 35                     :REM Gust Velocity (ft/s)
320 AU = 500                    :REM Uninstalled Avionics Weight (lbs)
330 LA = .9                     :REM taper ratio
340 KV = .007                   :REM Viscous Drag Coefficient estimated from a
                                     NACA 2214 airfoil
350 ARH = 5.2                   :REM Horizontal Tail Aspect Ratio
360 ARV = 1.33                  :REM Vertical Tail Aspect Ratio
365 ARC = 4.7
370 F = 1                       :REM Set Flag to 1
380 L = 70 : D = 4              :REM Length and Diameter of fuselage (ft)
390 INPUT "Output Data File Name [drive]:[path\filename]";F$
400 OPEN F$ FOR APPEND AS 1 LEN=2000
410 REM
420 REM  ***  data entry ***
430 REM
440 INPUT "Cl";CL
450 INPUT "Lower, Upper Mach Number and Increment (ML,MU,I)";M1,M2,I
460 INPUT "Payload Weight (lbs)";WP
470 INPUT "Thickness Ratio";TC
480 INPUT "Sweep Angle (radians)";SW
490 INPUT "Limit Load Factor";N
500 INPUT "Aspect Ratio";AR
520 INPUT "Number of Propellers";NP
530 INPUT "Number of Blades per Propeller";NB
540 INPUT "Diameter of Propeller";DP
550 DW = WP : WG=WP : WI = WP : REM weight increment
555 CLS
560 REM
570 REM *** begin iteration ***
580 REM
```

```
585 PRINT "Iterating Mach Number: "
590 FOR M = M1 TO M2 STEP I
596 LOCATE 1,25 : PRINT M;"          "
600 WS = .5*RO*SS^2*M^2*CL : REM wing loading
610 GOSUB 1100 : REM Calculate Weight
620 IF WG > 50000!   THEN 760 :  REM diverged
630 IF ABS(WI-WG) < 5 THEN 750 : REM check for convergence
640 DELTA = SGN (WI - WG)
650 IF F = 1 THEN ODELTA = DELTA
660 IF DELTA = ODELTA THEN LW=WI ELSE 710
670 F = 2
680 WI = WI + DW
690 ODELTA = DELTA
700 GOTO 600
710 DW = DW/2 : F = 1
720 WI = LW
730 DELTA = ODELTA
740 GOTO 600
750 GOSUB 810
760 WI = WP : DW = WP : F=1 : NEXT M
770 PRINT "Iteration complete....."
780 CLOSE 1
790 STOP
800 REM
810 REM *** print routine ***
820 REM
830 PRINT#1,M;CL;WG;S;AX;LD;OPT   : REM Output to data file
840 LPRINT "Component Weights at Mach Number";M;"and Cl=";CL;"T/C";TC;"AR=";AR
850 LPRINT
860 LPRINT "Wing Weight:                     ";WW
870 LPRINT "Weight Thin Film:                ";WA
880 LPRINT "Weight Ext Rectenna:             ";WX
890 LPRINT "Horizontal Tail Weight:          ";WT
900 LPRINT "Vertical Tail Weight:            ";WV
910 LPRINT "Controls Weight:                 ";WC
920 LPRINT "Fuselage Weight:                 ";WF
930 LPRINT "Landing Gear Weight:             ";WL
940 LPRINT "Hydraulic System Weight:         ";WH
950 LPRINT "Electrical System Weight:        ";WE
960 LPRINT "Motor Weight(Inst & Ind.)        ";WM
970 LPRINT"Gearbox Weight                  :";WTR
980 LPRINT "Propeller Weight:                ";WB
990 LPRINT "Payload Weight:                  ";WP
1000 LPRINT "Gross Weight:                    ";WG
1010 LPRINT "Initial Guess:                   ";WI
1020 LPRINT "Wing Area: ";S; "sq. ft."
1030 LPRINT "External Rectenna Area:";AX;"sq. ft."
1040 LPRINT "Wing Loading:";WS
1050 LPRINT "L/D optimum";OPT
1060 LPRINT "L/D Ratio:";LD
1070 LPRINT "Power Required:";PR
1080 LPRINT:LPRINT
1090 RETURN
1100 REM
1110 REM   *** weight calculations ***
1120 REM
1130 DR = 0
1140 GOSUB 1470 : REM calculated lift to drag ratio
1150 K1 = (WI * N  * 1.5) / (10^5)
1160 WZ =  ((K1^.65) * ((AR/COS(SW))^.57) *  (S/100)^.61)
```

```
1180 WW = .75 * (96.948 * (WZ * (((1+LA)/(2*TC))^.36)*1.224744)^.993)
1190 WT = .75 * ( 127*((K1^.87)*((.01*SH)^1.2)*1.3976*((ARH/(12*TC))^.5))^.458 )
1200 WV = .75 * ( 98.5*((K1^.87)*((.01*SV)^1.2)*((ARV/(12*TC))^.5)))
1210 WC = 1.08*((WI)^.7)
1220 WF = .75* ( 200 * (((K1)^.286)*((L/10)^.857)*((2*D)/10)*1.36302)^1.1)
1230 WL = .03 * WI
1240 WH = .0005*(WI^1.28) : REM hydraulic system weight
1250 WO = 2.117*(AU^.933)
1260 WE = 426*((WO/1000)^.51)
1270 PG = 1.22 * (WI/LD)*SS*M/550 :  REM Initial Power Guess
1280 WB = (31.92*NP*(NB^.391))*((DP*PG*.001)^.782) : REM prop weight
1290 AN = PG/.0872 : REM .0872 hp/ft2
1300 AX = AN - (WI/WS) : REM cover wing and horizontal tail
1310 IF AX <= 0 THEN 1360
1320 WX = AX * .2044 : REM .2044 lb/ft2 (film + structure)
1330 WA = (WI/WS) * .076 : REM weight of thin film and reflecting plane
                             on entire wing
1340 DR = (4*AX/3.14159)^.5
1350 GOTO 1380
1360 WA = AN * .076 : REM weight of thin film and reflecting plane on
                       portion of wing needed
1370 WX = 0
1380 WM = (2.575*(PG/NP)^.922)*NP
1390 WTR = .25 * PG
1400 GOSUB 1470
1410 PR = 1.22* (WI/LD) * SS * M / 550
1420 IF ABS (PG-PR) > 1 THEN 1270 : REM check for power required convergence
1440 WG = WV + WW + WT + WF +WE + WH +WP +WX +WA +WC+WL +WB + WTR +WM
1450 RETURN
1460 REM
1470 REM *** calc lift to drag ratio ***
1480 REM
1490 S = WI / WS
1500 SV = .1*S : SH = .2*S
1590 B = SQR( AR * S )                       : REM Span of Main Wing
1600 CBAR = S / B
1610 NRE = (RO*SS*M*CBAR) / MU
1620 CF=.074/((NRE)^.2)
1630 CW = CF * 2
1640 BH = SQR ( ARH * SH )                   : REM Span of Horizontal Tail
1650 BV = SQR ( ARV * SV )                   : REM Span of Vertical Tail
1660 NRE = (RO*SS*M*(SH/BH)) / MU
1670 CF = .074 / (NRE^.2)
1680 CH = CF * ( 2*SH ) / S
1690 NRE = (RO*SS*M*L) / MU
1700 CF=.074/((NRE)^.2)
1710 CU = CF * (L*3.14159*D)/S
1720 NRE = (RO*SS*M*(SV/BV)) / MU
1730 CF = .074 / (NRE^.2)
1740 CV = CF * ( 2*SV ) / S
1750 CR = 0
1760 IF DR = 0 THEN 1800
1770 NRE = (RO * SS * M * DR ) / MU
1780 CF = .074 / ((NRE^.2))
1790 CR = CF * (2*AX /S)
1800 CO = 1.25 * (CU + CW + CR + CH + CV)
1810 CD = CO + ((CL^2) / (3.14159 * AR * E )) + KV*CL^2
1840 RETURN
1850 END
```

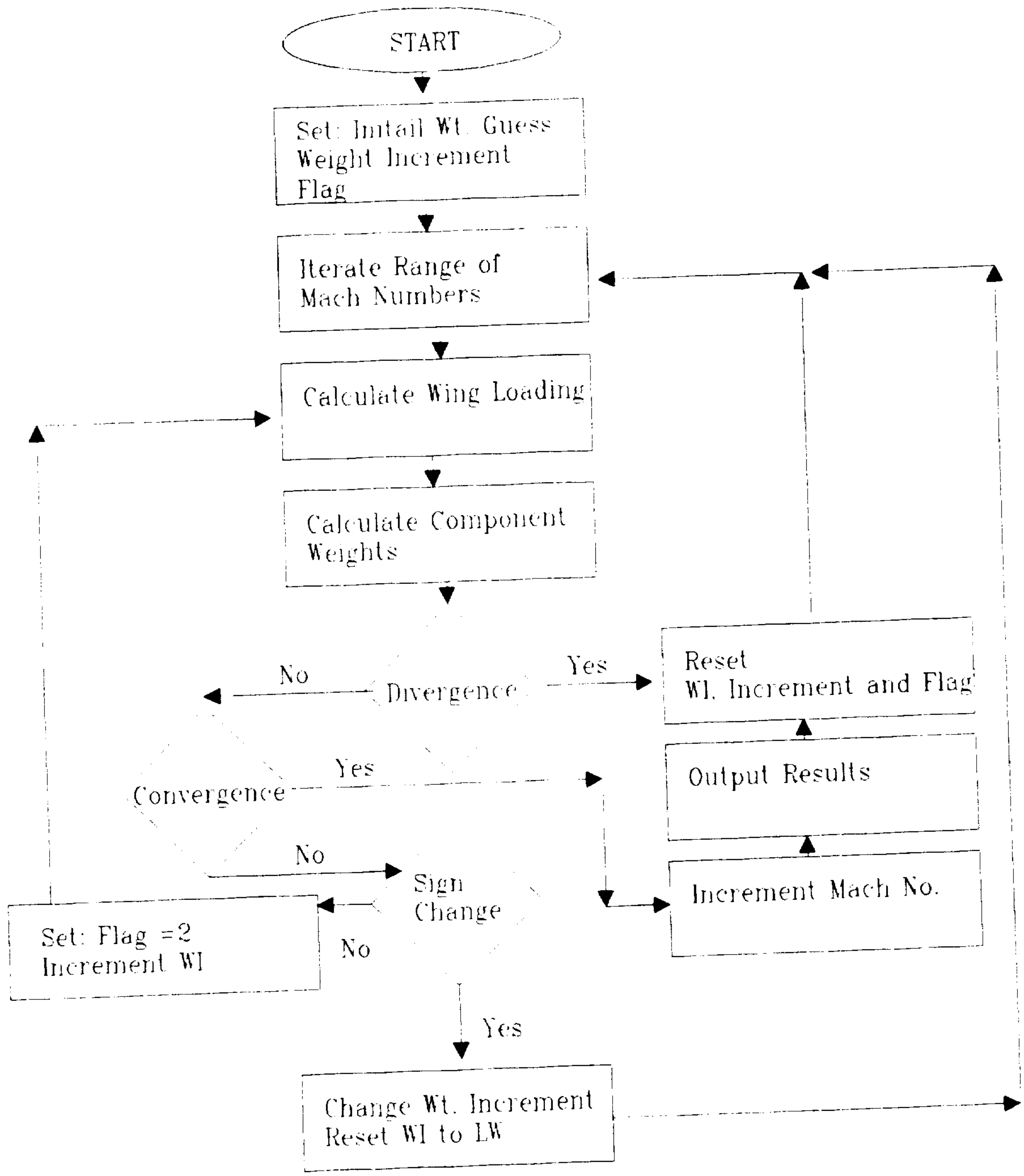
Figure A.3.1

First Iteration Weight Code Constants

Altitude	: 100,000 ft.
Air density	: 3.211(10^{-5}) lb/ft^3
Viscosity	: 3.15(10^{-7}) slug/(ft s)
Speed of sound	: 1003.2 ft/s
Span efficiency factor	: .85
Taper ratio	: .9
Viscous drag coefficient	: .007
Fuselage length	: 28 ft
Fuselage diameter	: 4 ft

Convergence Method Flow Chart

Figure A.3.2



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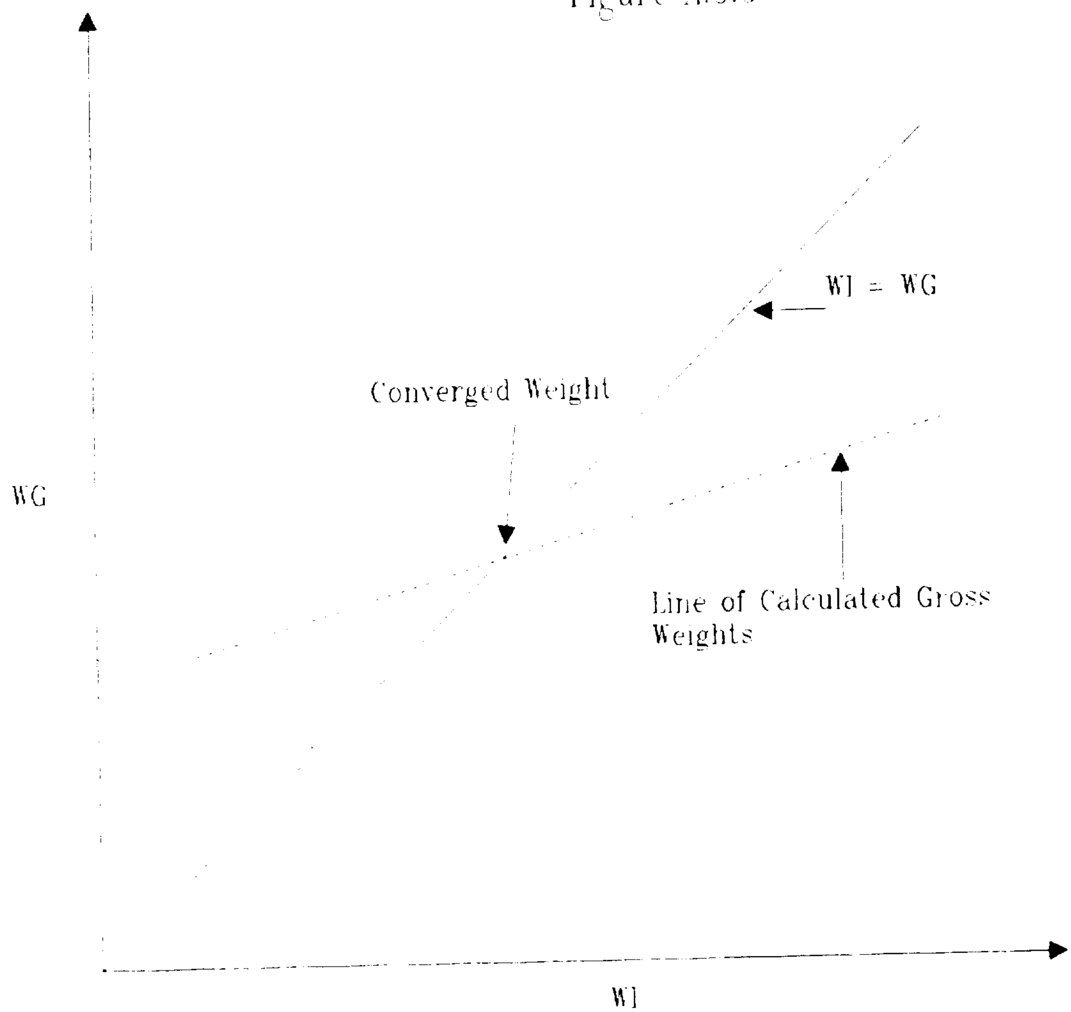
Graphical Convergence Example

Figure A.3.3



WI - Initial Weight Guess

WG - Iterated Gross Weight

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Appendix A.4

Theoretical Drag Calculation

The wing-body drag polar is determined from the methodology in Reference 24. A brief overview is given here.

$$C_D = C_{DO} + K' C_L^2 + K''(C_L - C_{LMIN})^2 \qquad \text{(A.4.1)}$$

where:

- C_{DO} - zero lift drag coefficient for wing-body (determined from C_L vs. α curve
- K'' - viscous drag factor
- K' - inviscid drag factor
- C_{LMIN} - lift coefficient at minimum drag may be approximated by lift coefficient at zero angle of attack

The zero lift drag coefficient term is comprised of the zero lift drag on the wing and of the fuselage. They are evaluated as follows:

$$C_{DO(wing)} = C_f[1+L(t/c)+100(t/c)^4]R\ S_{wet}/S_{ref} \qquad \text{(A.4.2)}$$

where:

- C_f - turbulent flat plate skin friction
- L - maximum thickness location factor
- t/c - maximum wing thickness
- R - lifting surface correlation factor
- S_{wet} - wetted wing area
- S_{ref} - reference area (wing planform)

Similarly, the drag on the body is:

$$C_{DO(body)} = C_f[1+60/(l_B/d)^3+.0025(l_B/d)]\ S_s/S_B \qquad \text{(A.4.3)}$$

where:

- l_B - length of the body
- d - diameter of body
- S_s - wetted area of body
- S_B - maximum cross sectional area

The value for the inviscid drag factor was also determined from Reference 24.

$$K' = \frac{1}{e\pi AR} \qquad (A.4.4)$$

where:

$$e = e'[1-(d/b)^2] \qquad (A.4.5)$$

d/b - body diameter to span ratio
e' - Weissinger wing planform efficiency factor

The value of K'' was determined from Reference 30.

After evaluation, the final equation for the wing-body drag polar reduces to :

$$C_D=.0154+.02167\ C_L^2+.1(C_L-C_{LMIN})^2 \qquad (A.4.6)$$

Appendix A.5

Static Structural Analysis

Preliminary Design

To begin the analysis, simple wing models consisting mainly of the wing box were used. The wing had a chord (c) of ten feet, a span (b) of 250 feet and an aspect ratio (AR) of twenty-five. To minimize the size of the computational model, only one - half of the wing was modelled due to symmetry. Translations in the X, Y, and Z directions were restrained at the aircraft centerline. Aerodynamic forces were represented by an elliptic load distribution of 5,000 pounds on each wing panel. The wing was constructed out of graphite - epoxy with a modulus of elasticity of 20 X 10^6 psi, a Poisson's ratio of 0.28 and a density of 2.91 slugs/ft^3.

After running this ANSYS model a maximum deflection of 2.9 feet, an average maximum shear stress of 0.271 ksi and an average equivalent stress of 114.6 ksi were obtained. The ultimate stress for graphite - epoxy is 69.9 ksi. The vertical reaction forces, 4,690 pounds, differ from the wing loading, 5,000 pounds, by only six percent. This difference in loading is due to the fact that the pressure was placed perpendicularly to the surface of the upper panel, which is inclined relative to the wing refrence panel, hence the vertical force component differs by the cosine of the angle.

These results indicate that the wing will fail due to the high equivalent stresses. Furthermore, the weight

of each wing panel is 5,308.7 pounds, which exceeds the wing weight allowance of 2700 lbs.

Next, it was necessary to expand the structural model to include both leading and trailing edges, thereby making the model more realistic. Second, the thickness of the wing material was decreased in order to reduce to overall weight of the wing. The thickness of the first four sections out from the centerline was 0.1 inches, of the middle four sections 0.01 inches, and of the last four sections 0.0025 inches. The same material properties were used in each section.

The large decrease in material thickness, with a commensurately large decrease in cross sectional area, resulted in a dramatic increase in stresses every fourth section. For this configuration the stresses obtained were 11.8 ksi for the average maximum shear stress and 206.25 ksi for the average maximum equivalent stress, as shown in Figures A.5.1 - A.5.5.

The stresses proved to be three times as much as the ultimate stress, 69.9 ksi. Also, the minimum thickness of graphite - epoxy is approximately 0.03 inches as stated in Reference 28.

To alleviate the above conditions the panel thickness was reduced gradually, which yielded a gradual change in the stress distribution. Second, to reduce the maximum stresses flanges were added to the wing spars, and not just webs as previously shown.

Finally, in order to have a more realistic model, a more accurate elliptical loading distribution which correctly represents the equivalent bending moment, was input. This will result in the correct I beam thicknesses and cross sectional areas.

The equation of the spanwise elliptical loading is

$$W(y)= Wo(1-(y/(b/2))2))^{\frac{1}{2}},$$

where Wo is the maximum load (found at the centerline), b is the span of the wing, y is the distance along the span and W(y) is the load in lbs/ft found at varying points along the wing.

The bending moment is found by integrating the above equation twice. To accomplish this task a program was written to perform the integration, as shown at the end of this Appendix. To test the accuracy of the program the equation of elliptical loading was changed to an equation of uniform load and then later to an equation of load concentrated at one point. All three runs were found to be accurate, as shown in Figures A.5.6 - A.5.14.

Model#1

After completion of this preliminary model and analysis, new wing geometry and weight was provided by the aerodynamics group. The aspect ratio (AR) was 15, the thickness to chord ratio was 0.12 and the total wing weight was 2704.644 pounds.

Given the total wing area to be 3633.8 square feet and the minimum thickness of graphite - epoxy to be 0.035 inches, the skin weight would be 2270 pounds. The skin weight left an unreasonably low weight of 434 pounds available for use in interior structural support.

Due to this high weight, it became necessary to turn to another material to construct the skin. Aluminum was chosen because it is possible to reduce the material thickness to 0.01 inches. But with aluminum the density rises to 5.42 slugs/ft^3 yielding a skin weight of 1059 pounds. This left 1645.6 pounds to be used for the interior structural support. Next, it was necessary to determine the respective sizes of the two I beams in the wing. The forward wing spar was sized using 67 percent of the load, and the rear spar was size using 33 percent, as shown in Figures A.5.15 - A.5.20. For this initial analysis it was determined that the plane would be designed for an ultimate load factor of three, which later proved to be much lower than required.

Given the weight of the plane, from the

aerodynamics group, to be 6718 pounds and a load factor of n_{ult} = 3, the load the wings were designed to withstand was 20,154 pounds.

To get the maximum Wo one can use the relationship

$$L = (\pi * Wo * (b/2))/2.$$

Solving for Wo yields,

$$Wo = (L*4)/(\pi * b)$$

where L is the load of 20,154 lbs and b is the span of 233 ft. Hence, the final result is a Wo of 110.13 lb/ft. As stated previously the front wing spar carries 67 percent of the load and the rear wing spar carries 33 percent of the load. Hence, one gets a $Wo_{(front)}$ of 6.15 lb/in and a $Wo_{(rear)}$ of 3.03 lb/in.

By using the same integration program to integrate the elliptic load function,

$$W(y) = Wo(1-(y/(b/2))2)^{\frac{1}{2}},$$

one gets the moments on the front and rear wing spars. Additionally, from a Lotus 1-2-3 output one gets the average moment every 11.5 feet (in other words every span section).

To find the thickness of the front wing spar one

can use the following equation,

$$S_{min} = (M)_{max}/(\sigma)_{all}$$

In the above equation S_{min} is the minimum section modulus, $(M)_{max}$ is the maximum moment and $(\sigma)_{all}$ is the allowable tensile stress. Knowing that S=I/c, where c is the maximum distance from the neutral axis, and assuming h = b = 0.12*(chord) one can obtain the equation for the moment of inertia,

$$I_{xx} = (b*h^3/12) - (2*b_1*h_1^3/12)$$

The same procedure is used to find the thickness of the rear I beam. The results are given in Figures A.5.21 & A.5.22.

Nonlinear deflections

Using Euler - Bernoulli analysis from ANSYS one gets a vertical deflection of 198 inches. Now using Figure 2 on page 1053 of Reference 17 a deflection to length ratio d/l can be calculated in order to compare the results to a nonlinear deflection. Hence, one gets d/l equal to 0.14, K equal to 1.44 and t equal to 0.1 inches. Here K is equivalent to

From the graph on page 1053 the Euler - Bernoulli theory shows a close relation to the Rhode's method in this case. However, one should note that equation (A.5.1) is valid for a uniform load distribution and not for an elliptical distribution, hence the validity of its use in this case is uncertain. Also, one should note that the Rhode's method of solving nonlinear deflections was the only one practical.

Preliminary Loads Analysis:

The first step in finding the correct load distribution along the wing is to find the proper airfoil section lift coefficients along the span of the wing. Given that the lift over the wing is elliptical, one can use the equation $W(y) = Wo(1-y/(b/2)^2))^{\frac{1}{2}}$. Also given that $W(y) = CL*0.5*den*V*V*c_{(y)}*y$, one can equate the above equations to solve for Cl. The following conditions were used, a Wo of 110.14 lb/ft, a semi-span of 115 ft, a density of 3.211E-5 slugs/ft^3, a velocity of 603.2 ft/s, a mean chord c(y) of 15 ft and a span section of 11.5 feet. The final equation will then be

$$CL = \frac{2*(Wo(1-(y/(b/2))\hat{}2)^{\frac{1}{2}})}{(den*V*V*c(y))} \qquad (A.5.2)$$

As one moves out from the root to the tip of the wing the lift coefficient varies with Wo, and Wo varies elliptically.

After finding Cl along 10 evenly spaced positions along the span the lift coefficients were given to the aerodynamics group. The aerodynamics group then calculated the corresponding pressure coefficients Cp chordwise along each of the Cl's given.

However, this analysis (load = 3W) led to a Cl of 1.8 (the aerodynamic group's CL = 0.6 for L=W). Knowing that the plane would stall at about a CL of 1.0 the situation had to be reevaluated. It was decided to solve for chordwise pressure coefficients across the span. Hence, one had to assume that the maximum CL encountered during gust conditions would be a CL = 1.0. But this would give a different velocity than that used by the aerodynamics group (M=0.44). Hence, to get the correct wing loading of 20154 lbs a new velocity was required, which turned out to be 603.2 ft/s. Knowing the velocity one can then calculate q (dynamic pressure) and Cl along the span to determine the pressures on the wing. The corresponding results are given in Figure A.5.24.

Using the average panel pressures the results were entered into the ANSYS model with a gravity load, the results were as follows: vertical deflection of 14 feet, maximum equivalent stress of 30 ksi and maximum shear of 733 psi. One analysis which has yet to be considered is the effects due to torsion. Since ANSYS has no direct commands to check for torsion it was determined if the torsion in wing was too high by performing a twist analysis.

Another note of interest is that the deflection in the chordwise direction was 7.33 inches, which is believed to be too high for such a model. This deflection could be due to the low stiffening in the

model. One way to compensate for the lack of stiffening in the chordwise direction was to replace the bar elements along the leading and trailing edges with beam element, since beam elements can better represent the stiffening components along the desired directions. The beams along the leading and trailing edges were made bigger (using the average real constants from the front and rear I beams respectively), which increased the weight to 2319 lbs.

The bar and beam elements along the leading and trailing edges were checked for twist. The desired angle of twist was plotted versus the semi-span of the wing. In order to determine the angle one must take the arcsine of the change in chordwise direction divided it by the chord. Hence one gets

$$\sin(x) = \text{delY}/\text{chord} \quad \text{(A.5.3)}$$

where delY is the change in displacement from the trailing to leading edge.

One can readily see from Figures A.5.25 & A.5.26 that the angle of twist reaches a maximum of 0.17 degrees. This means that the wing is experiencing very little twist and hence the static torsion on the wing is also very small.

Second Model

Since the first model seemed to lack sufficient stiffness a second model was developed. The geometry will be maintained from the previous model but with an additional I beam placed in the middle of the wing. Again, the I beams will vary in thickness as shown in Figures 6.1.13 - 6.1.16. The old I beams (wing spars) had unrealistic dimensions for their flanges, which were 21.6 and 14.4 inches wide. The flanges were redesigned and reduced to a width of 4.32 inches in the front, 3.74 inches at the middle, and 3.17 inches in the rear, but the cross sectional area was kept constant, as shown in figure A.5.27. There will also be twice as many span stations, 5.75 feet apart. Finally, updated pressure distributions supplied by the aerodynamics group have been used to calculate the applied load distribution on the wing.

Loads Analysis II

From the aerodynamics group's analysis of the chordwise pressure distribution one can establish a better chord and spanwise pressure distribution per panel on the new wing (160 surface panels).

The chordwise pressure coefficient distribution for a CL of 0.743913 was provided by the aerodynamics group. The wing structural model had four chordwise panels, so the pressure distribution was approximated as shown in Figure 6.1.2.

To determine if these are the correct pressures one can compare the calculated to the expected force and see how they relate. The comparison between the expected and calculated results are shown in Figure A.5.28.

Second Model

As mentioned in Reference 28, graphite - epoxy will not bond directly to aluminum. Graphite - epoxy will need an E glass filling between all joints where it is connected to aluminum. This will raise the weight of the wings significantly beyond the designated 2700 lbs.

The wing weight of the new model with aluminum skin and no E glass filling is 2800 lbs. The weight is already 100 lbs over the designated allowance without the weight of the E glass taken into account.

Further investigation on graphite - epoxy wing construction indicated that building a wing skin of 0.01 inches is possible. Graphite - epoxy can be manufactured to about 0.02 inches, so, as stated in Reference 26, it might be possible to produce graphite - epoxy to 0.01 inches in the near future. Hence, with this consideration in mind and the fact that this is still a first weight estimation, the skin was changed from aluminum to graphite - epoxy, reducing the wing weight to 2319 lbs.

Gravity Loading Analysis

With gravity alone, one gets the following results: a maximum vertical deflection of -1.45 feet, an average maximum equivalent stress of 5183 psi and a maximum horizontal displacement of -0.81 inches, as shown in Figures A.5.29 - A.5.34.

Pressure Loading Analysis

With just a pressure loading, the average maximum shear stress is -2.0 ksi, the average maximum equivalent stress is 39.7 ksi, the maximum vertical displacement is 10.9 feet, and the maximum horizontal (x) displacement is 6 inches, as shown in Figures A.5.35 - A.5.44. Comparing the results to those taken on the model which had only 10 span stations, larger flanges in the I beams and no middle I beam, the results were as expected. Since there was more interior stiffening in the model, the deflections were reduced from 16.5 to 10.9 feet (vertical direction) and from 7.33 to 6 inches (horizontal direction). The shear stresses were reduced from 2184 to 2005 psi. However, the average maximum equivalent stress increased from 33,153 to 42,138 psi. This increase is attributed to the width reduction of the I beam flanges from 21.6 to 4.32 inches. But, since graphite - epoxy's ultimate principle stress is 69.9

ksi, the wing is still acceptable.

```
1 REM Program #1
2 REM  This program allows you to take the double integral of an equation
10 REM Elliptical load distribution
20 OPEN "a:amoment.dat" FOR OUTPUT AS 1 LEN=2000
30 INPUT "what is the Max X value"; B
40 INPUT "how many sections"; NTOT
50 DIM LOD(NTOT-1),Y(NTOT-1),SHR(NTOT-1),MOM(NTOT-1)
60 DELY=B/NTOT
70 LOD(I)=0
80 Y(I)=0
90 X=0:GOSUB 1000:LOD(1)=Y
100 FOR I=1 TO NTOT
110 Y(I)=(I-1)*DELY
120 X=Y(I)
130 GOSUB 1000
140 LOD(I)=Y
150 NEXT I
160 Y(NTOT-1)=B
170 LOD(NTOT-1)=0
180 REM compute shear
190 SHR(NTOT-1)=0
200 FOR K=NTOT TO 1 STEP -1
210 LAVE=(LOD(K)-LOD(K-1))/2
220 SHR(K)=SHR(K-1)-(LAVE*DELY)
230 NEXT K
240 REM compute moment
250 MOM(NTOT-1)=0
260 FOR K=NTOT TO 1 STEP -1
270 MAVE=(SHR(K)-SHR(K-1))/2
280 MOM(K)=MOM(K-1)-(MAVE*DELY)
290 NEXT K
300 FOR X=NTOT-1 TO 1 STEP -1
310 PRINT#1,Y(X);LOD(X);SHR(X);MOM(X)
320 NEXT X
330 END
1000 IF X>B THEN X=B
1001 Y=6.15*((1-(X/B)^2)^(1/2))
1002 REM Put the equation you wish to integrate in the above line
1010 RETURN
```

```
1 REM Program #1
2 REM  This program allows you to take the double integral of an equation
10 REM Elliptical load distribution
20 OPEN "a:amoment.dat" FOR OUTPUT AS 1 LEN=2000
30 INPUT "what is the Max X value"; B
40 INPUT "how many sections"; NTOT
50 DIM LOD(NTOT-1),Y(NTOT-1),SHR(NTOT+1),MOM(NTOT+1)
60 DELY=B/NTOT
70 LOD(I)=0
80 Y(I)=0
90 X=0:GOSUB 1000:LOD(1)=Y
100 FOR I=1 TO NTOT
110 Y(I)=(I-1)*DELY
120 X=Y(I)
130 GOSUB 1000
140 LOD(I)=Y
150 NEXT I
160 Y(NTOT+1)=B
170 LOD(NTOT-1)=0
180 REM compute shear
190 SHR(NTOT+1)=0
200 FOR K=NTOT TO 1 STEP -1
210 LAVE=(LOD(K)+LOD(K-1))/2
220 SHR(K)=SHR(K+1)-(LAVE*DELY)
230 NEXT K
240 REM compute moment
250 MOM(NTOT-1)=0
260 FOR K=NTOT TO 1 STEP -1
270 MAVE=(SHR(K)+SHR(K-1))/2
280 MOM(K)=MOM(K+1)-(MAVE*DELY)
290 NEXT K
300 FOR X=NTOT-1 TO 1 STEP -1
310 PRINT#1,Y(X);LOD(X);SHR(X);MOM(X)
320 NEXT X
330 END
1000 IF X>B THEN X=B
1001 Y=6.15*((1-(X/B)^2)^(1/2))
1002 REM Put the equation you wish to integrate in the above line
1010 RETURN
```

Figure A.5.1
Deflection 47.15 inches

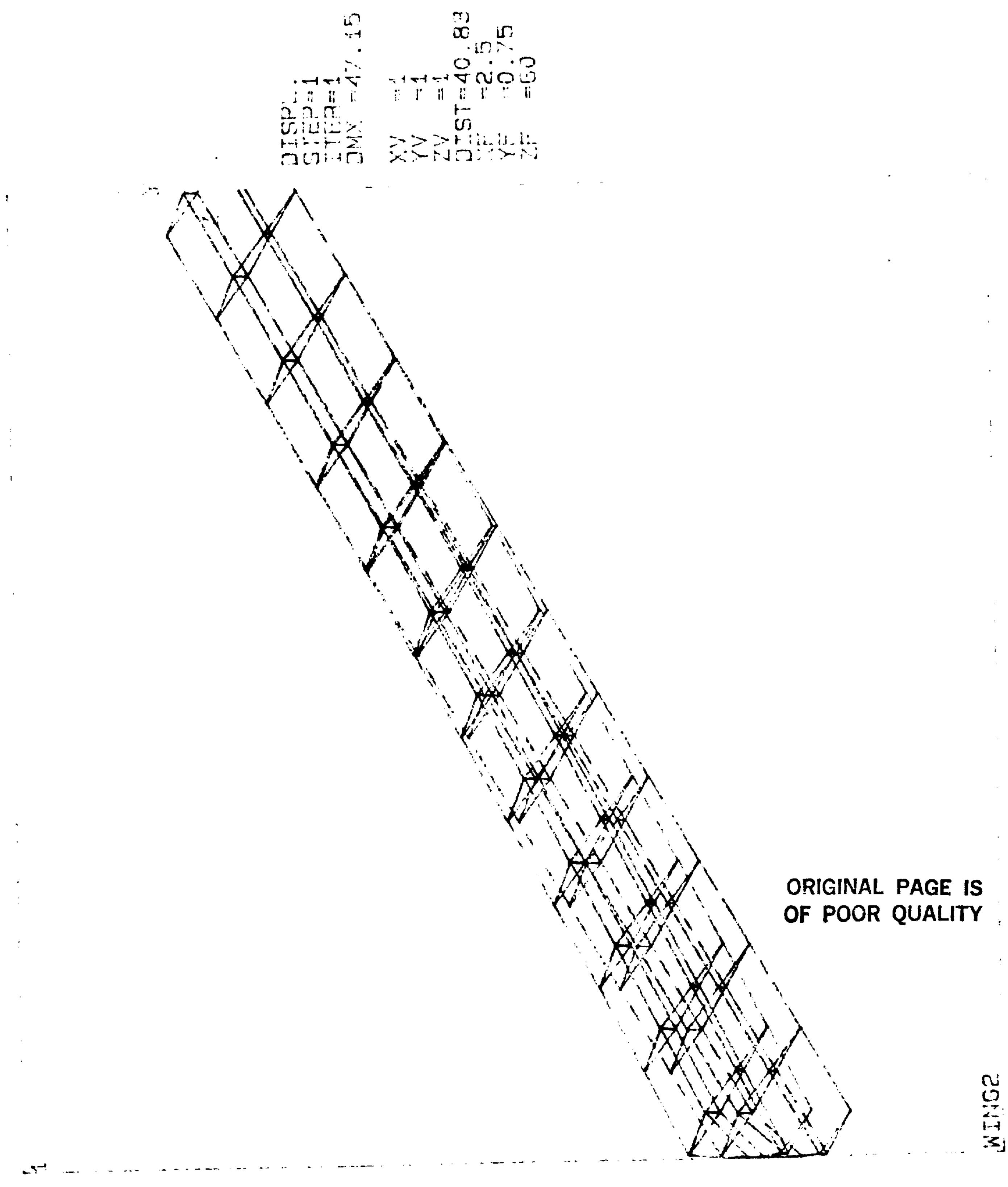

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Figure A.5.2
Shear Stress

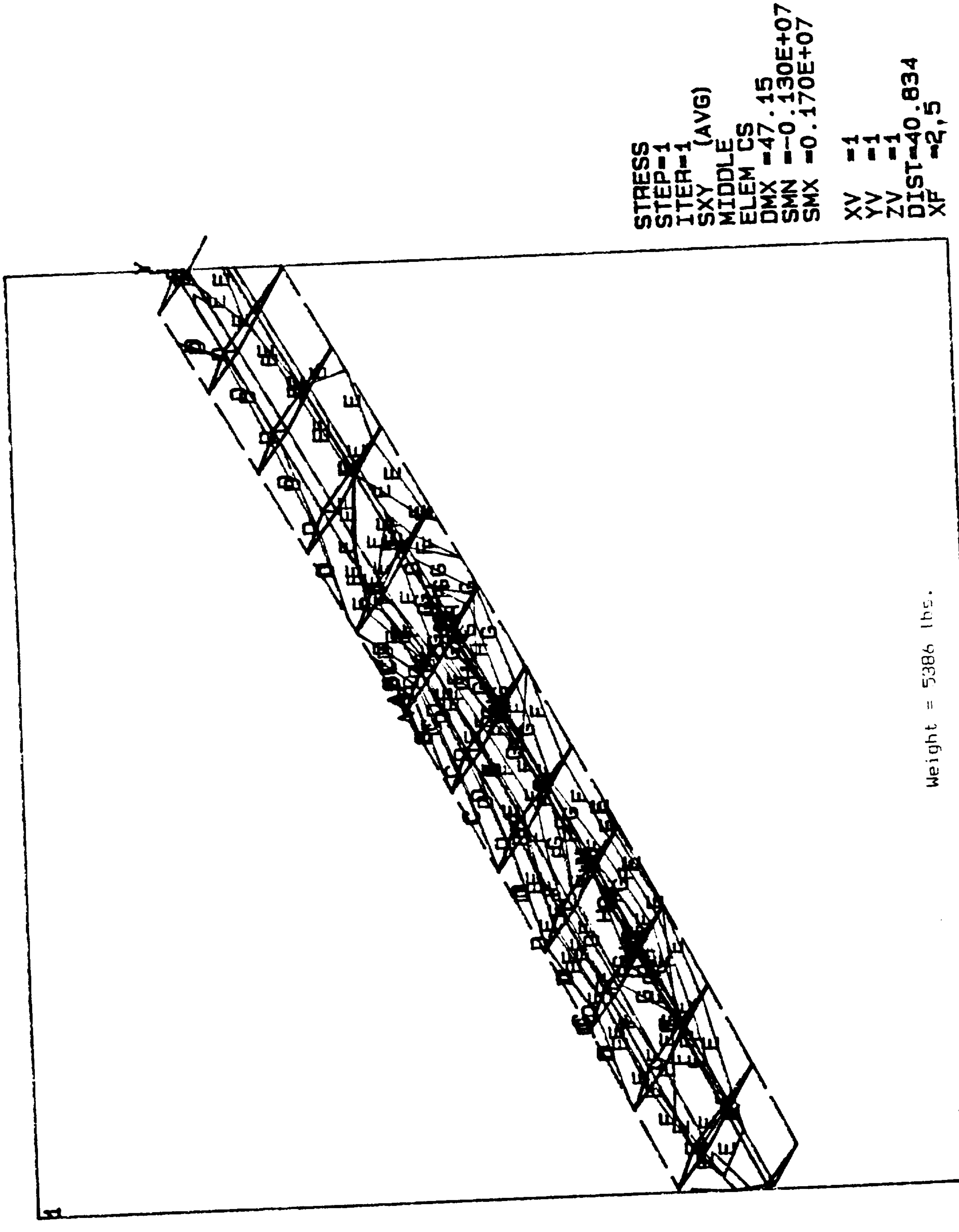
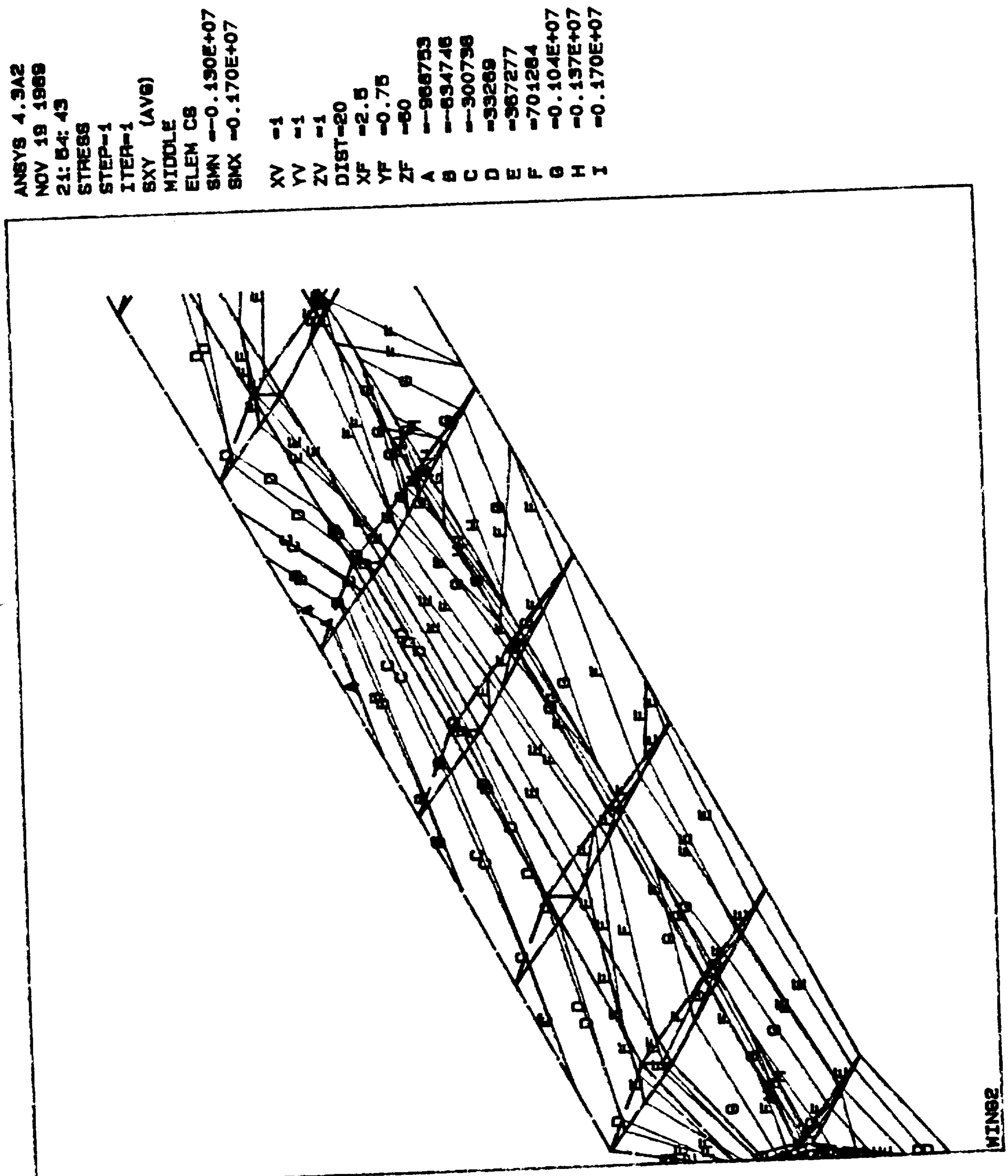


Figure A.5.3
Shear Stress



Weight = 5386 lbs.

Figure A.5.4
Equivalent Principle Stress

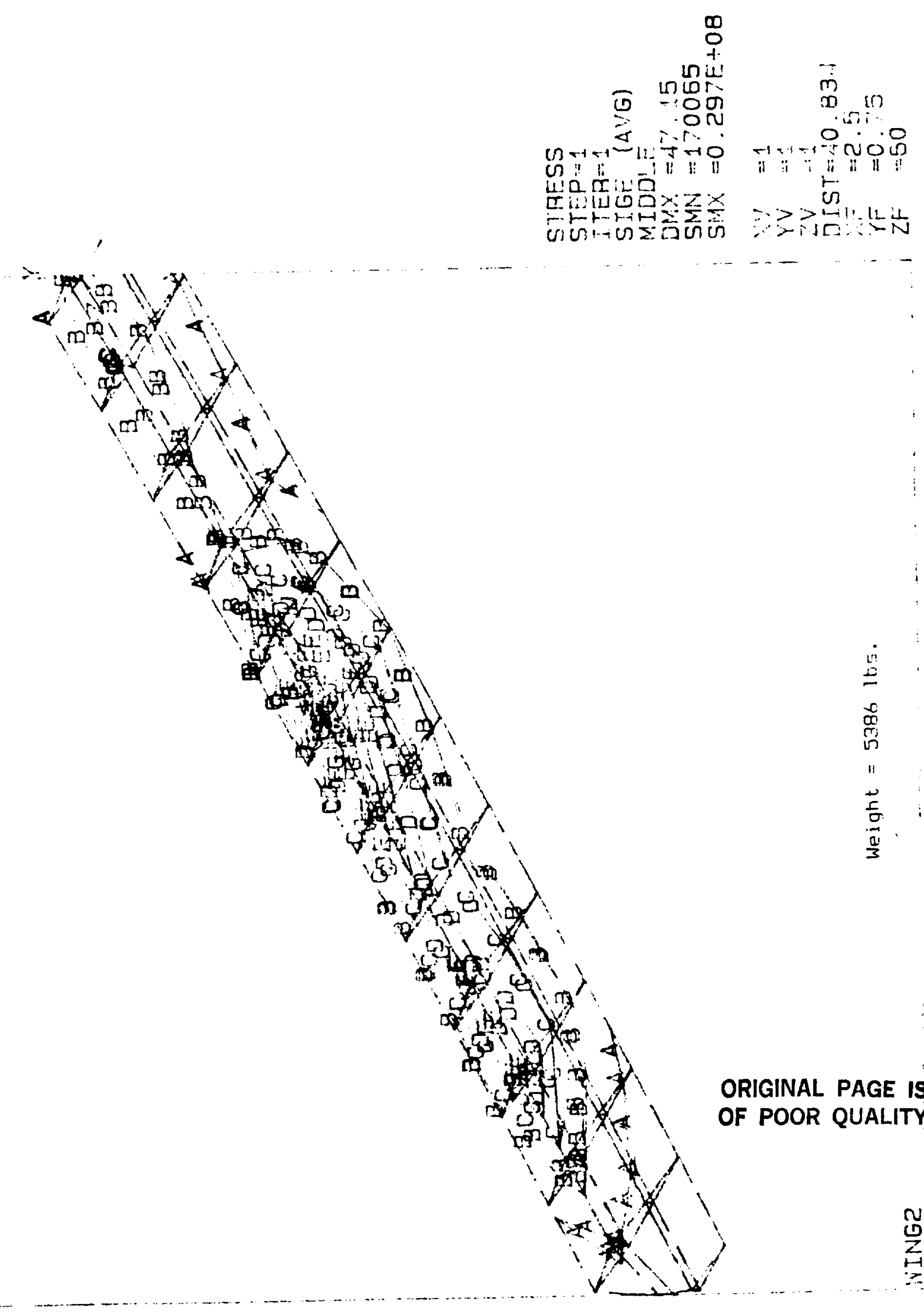


Figure A.5.5
Equivalent Principle Stress

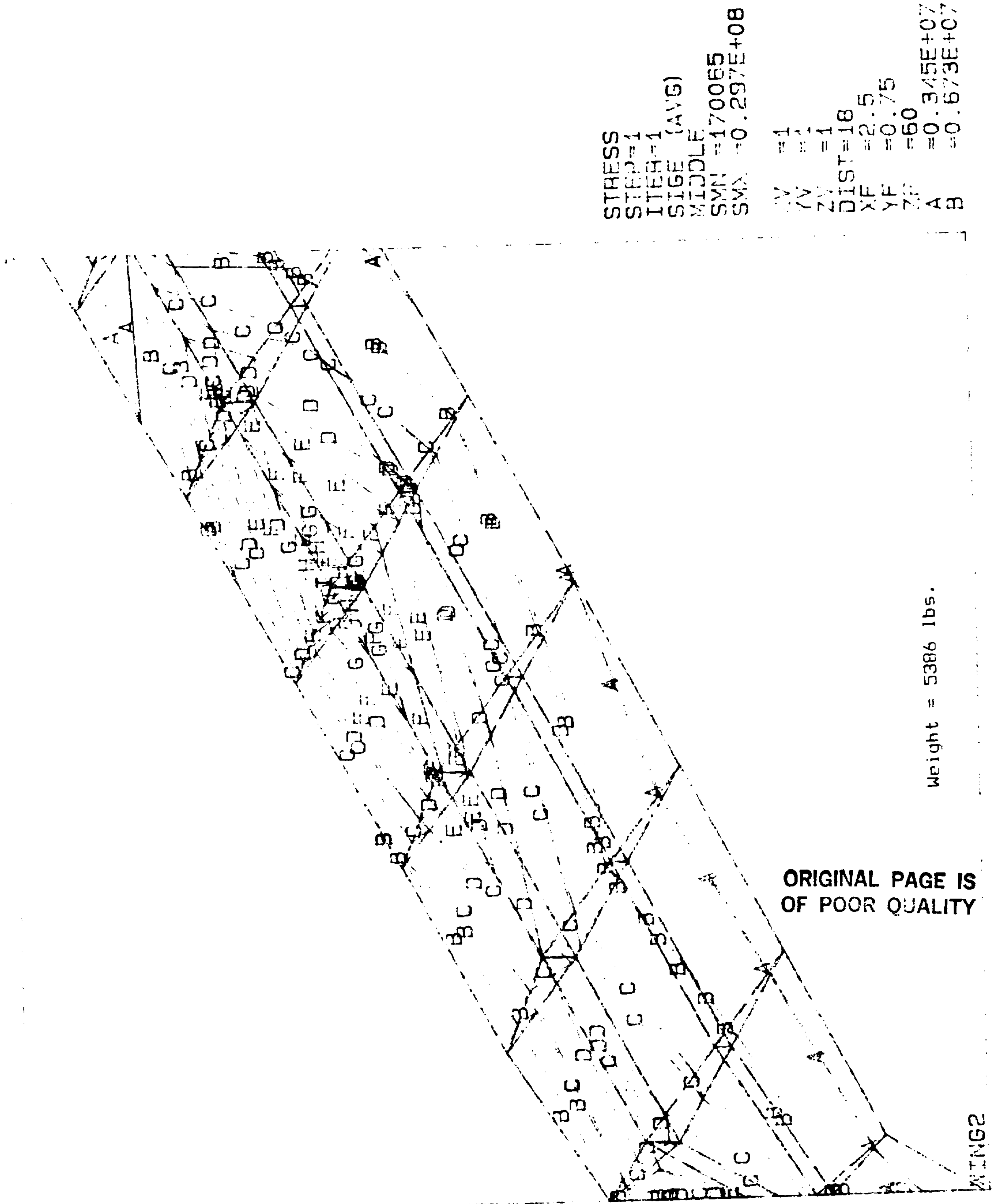


Figure A.5.6

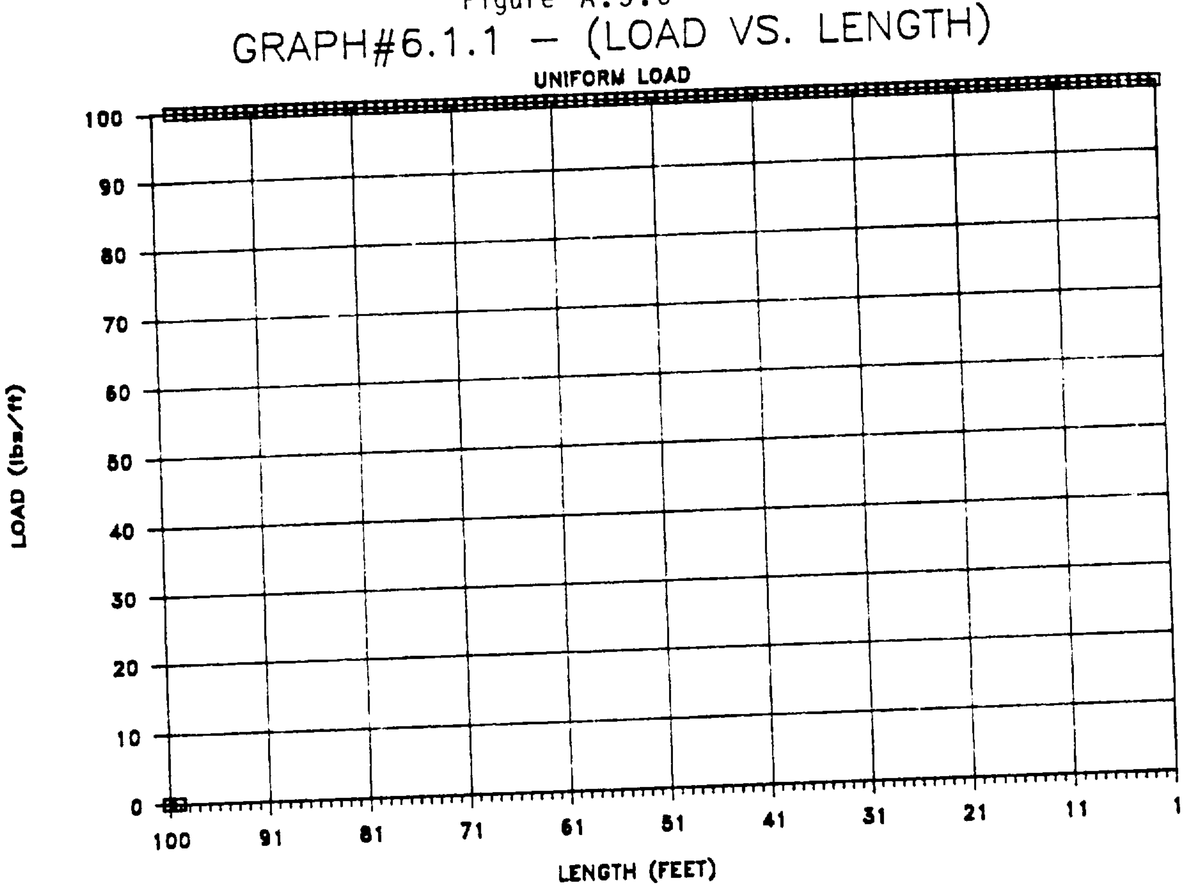


Figure A.5.7

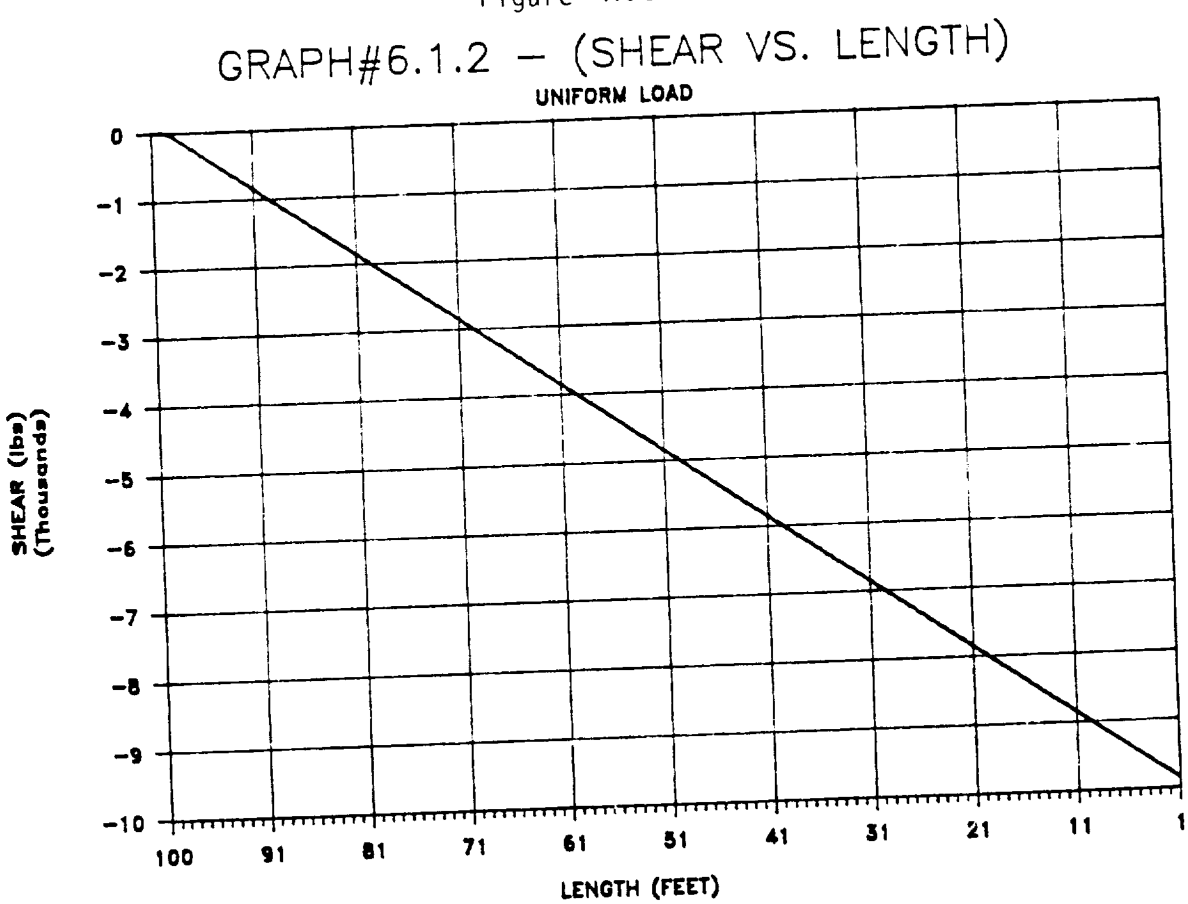


Figure A.5.8

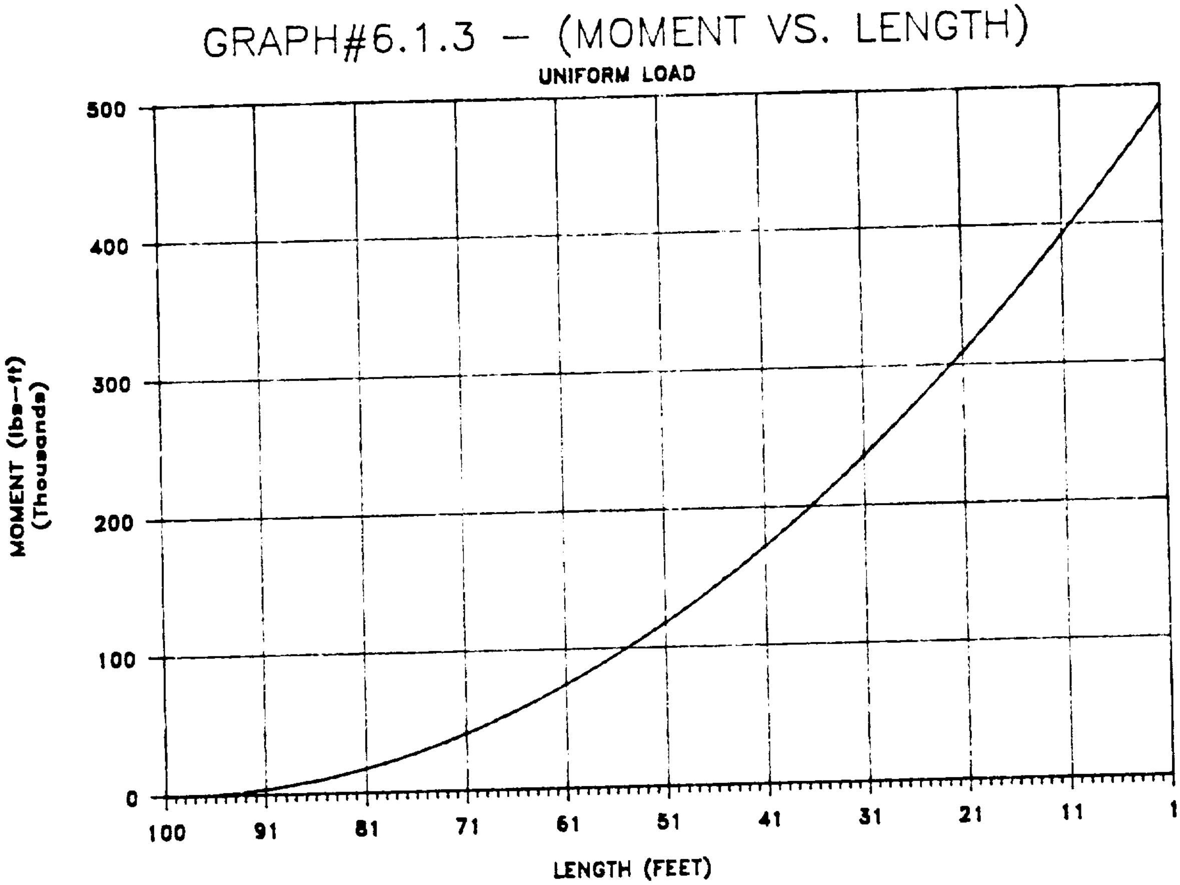


Figure A.5.9

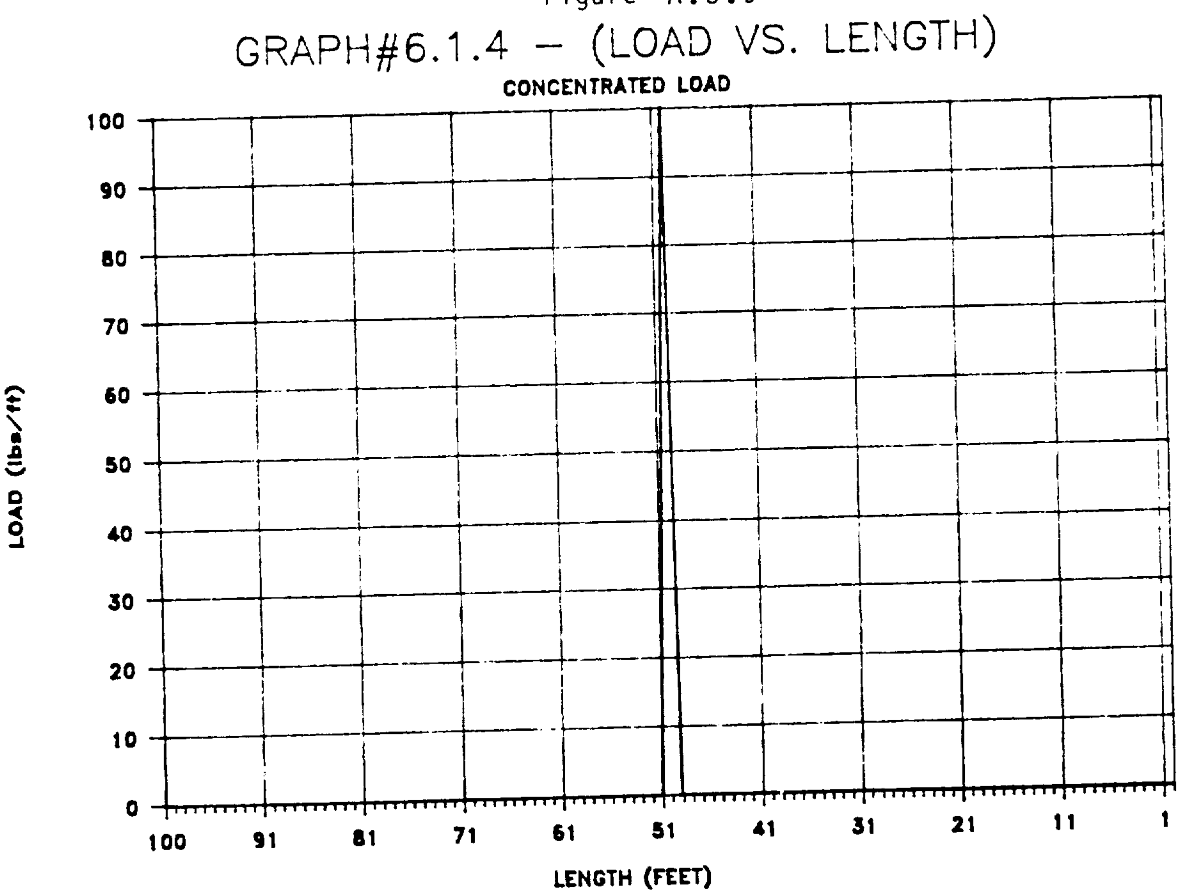


Figure A.5.10

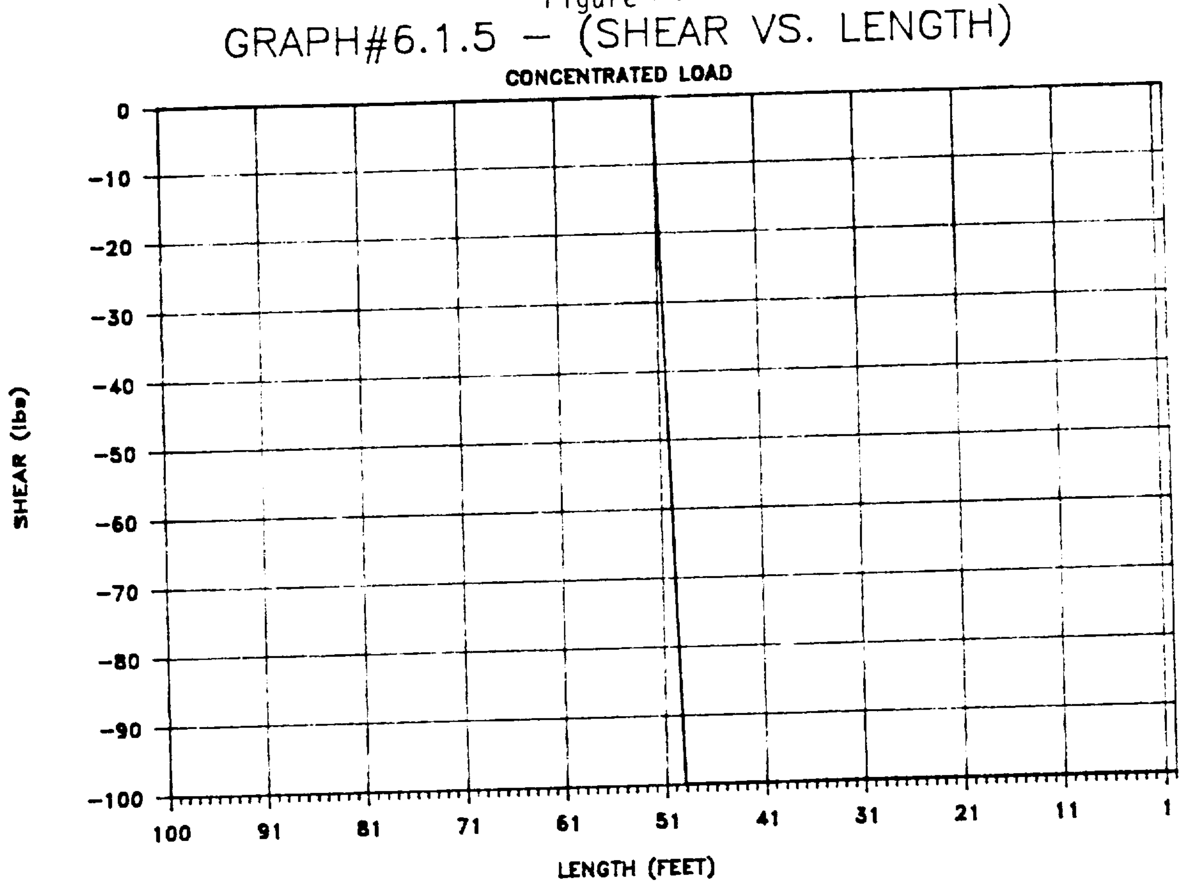


Figure A.5.11

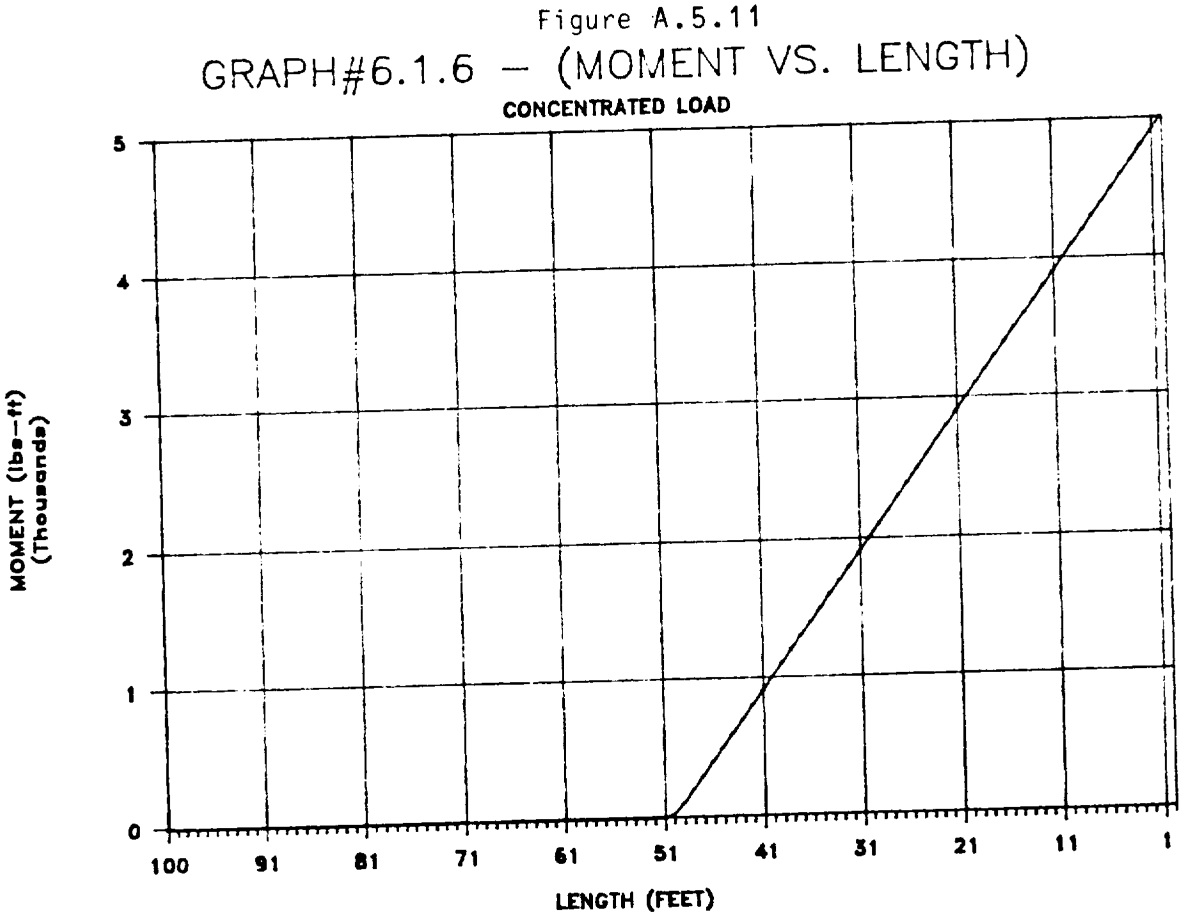


Figure A.5.12

GRAPH#6.1.7 — (LOAD VS.SEMI-SPAN)

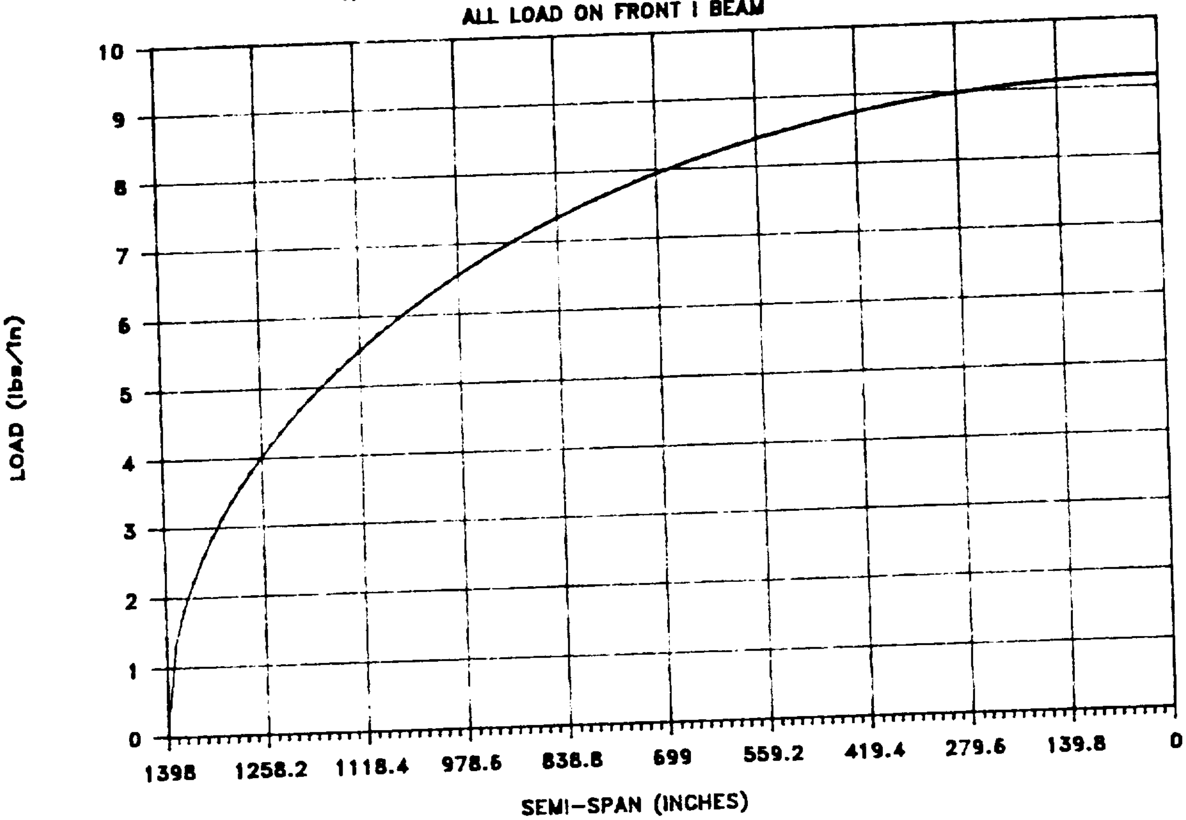


Figure A.5.13

GRAPH#6.1.8 — (SHEAR VS.SEMI-SPAN)

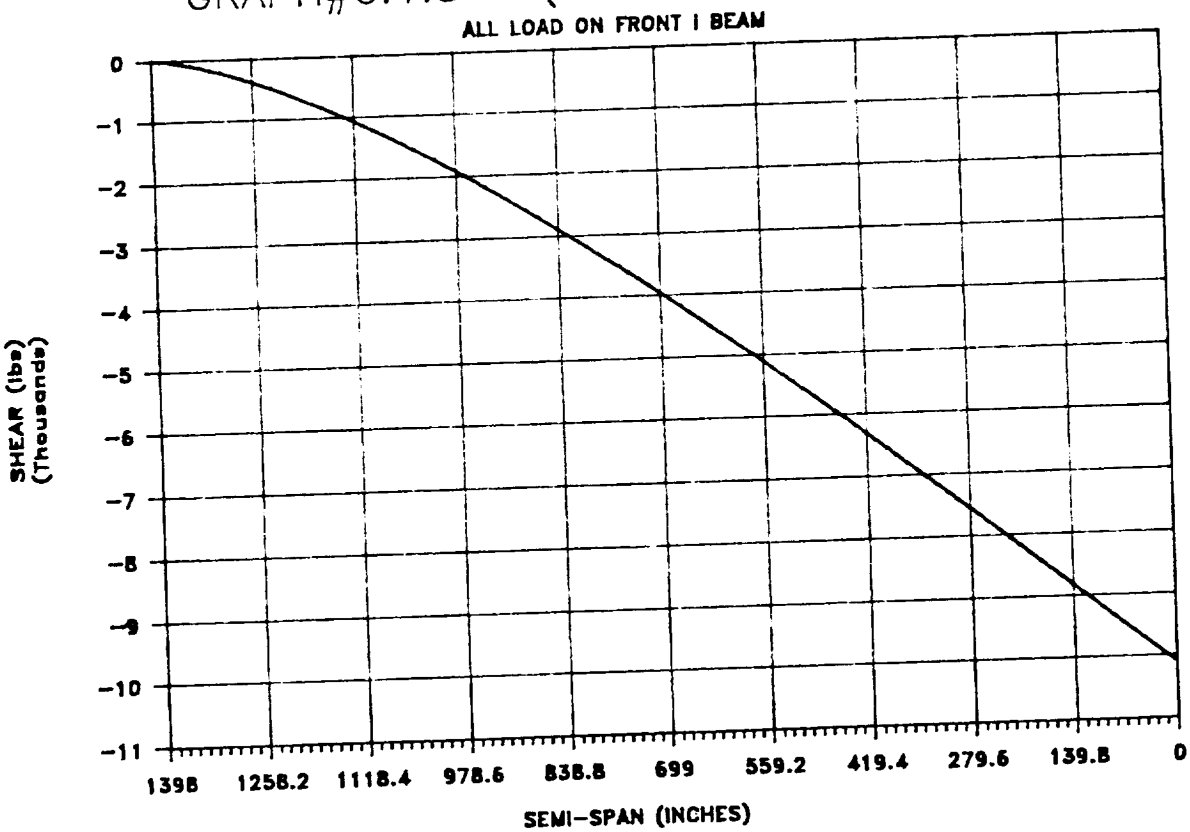


Figure A.5.14

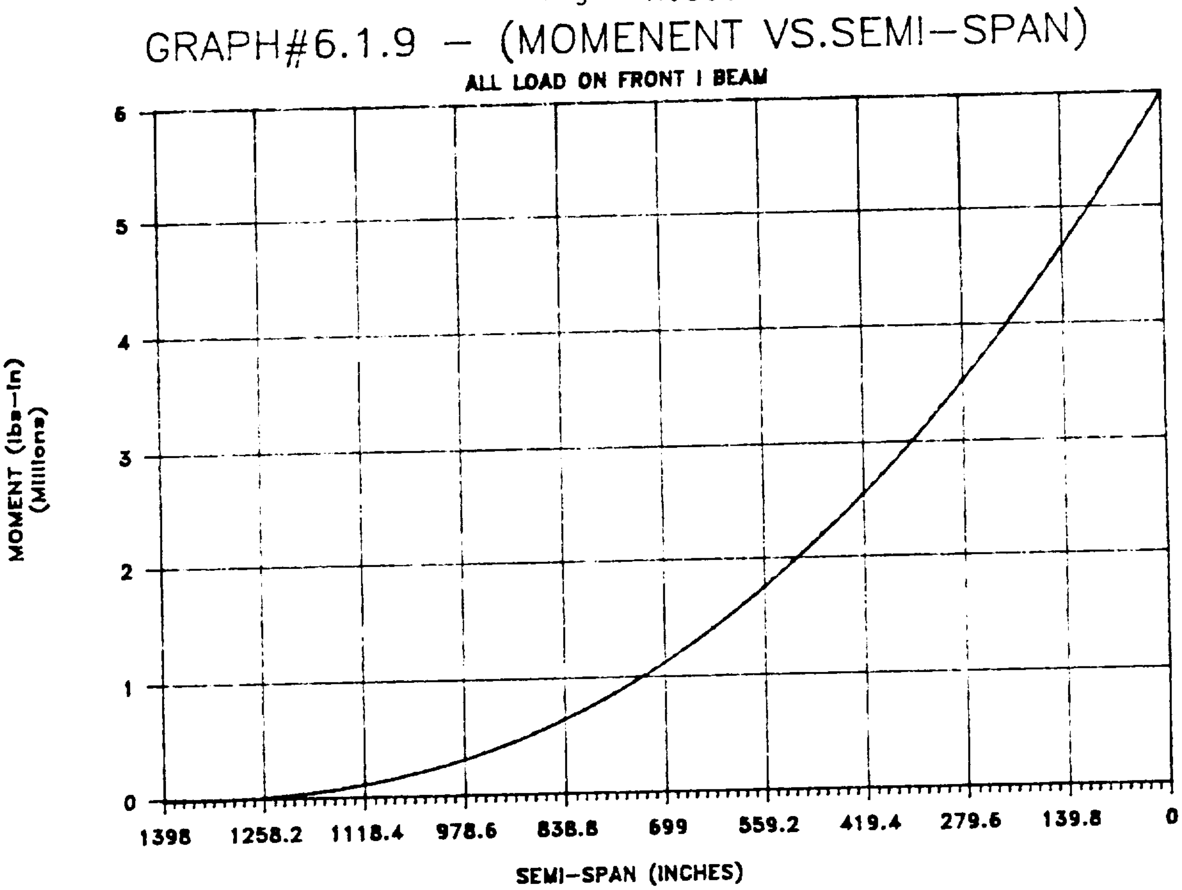


Figure A.5.15

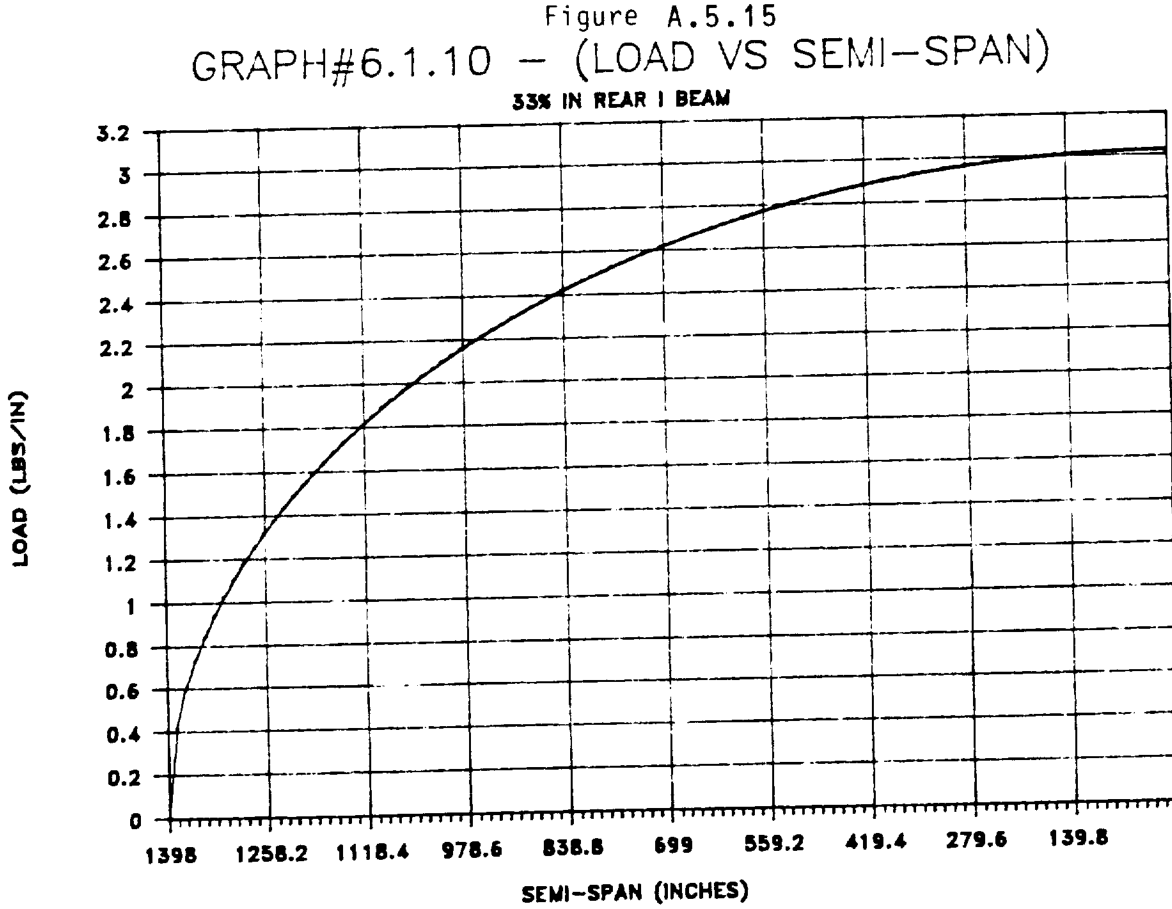


Figure A.5.16

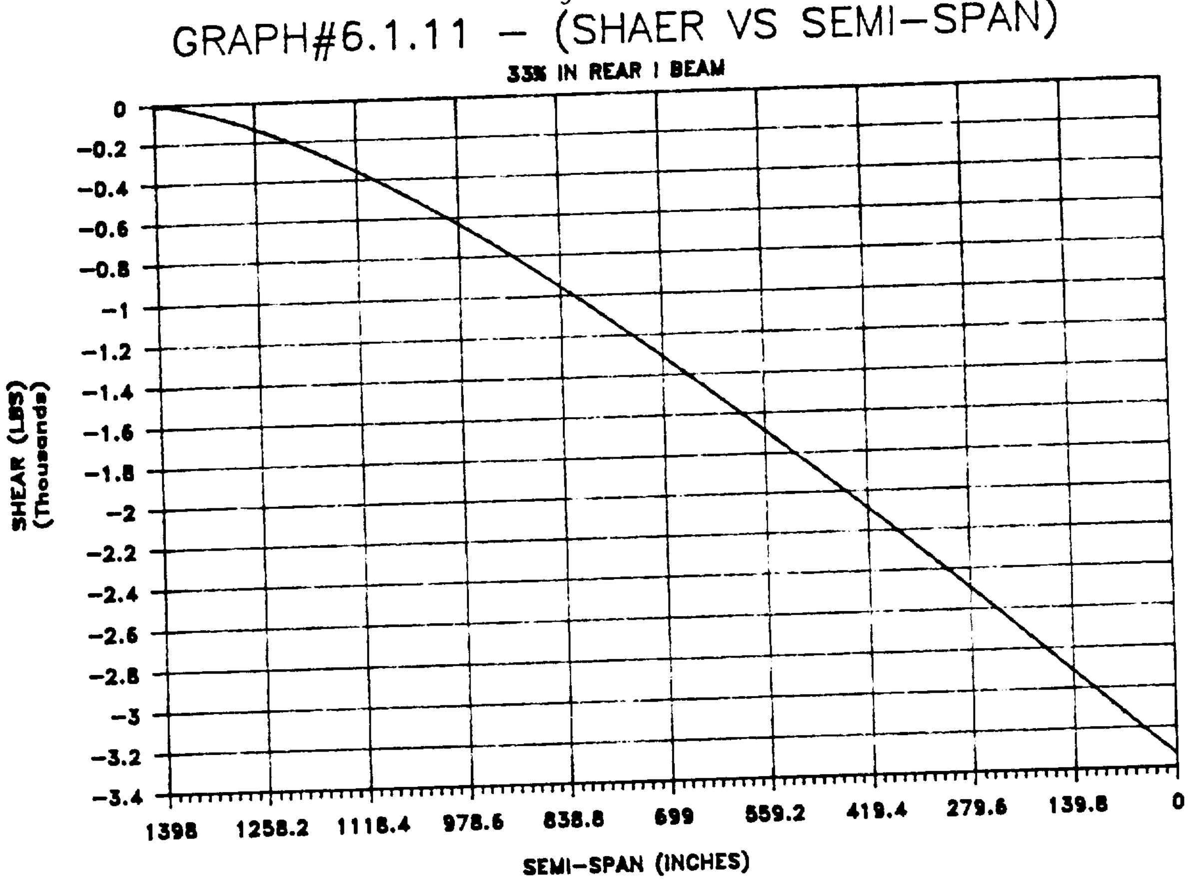


Figure A.5.17

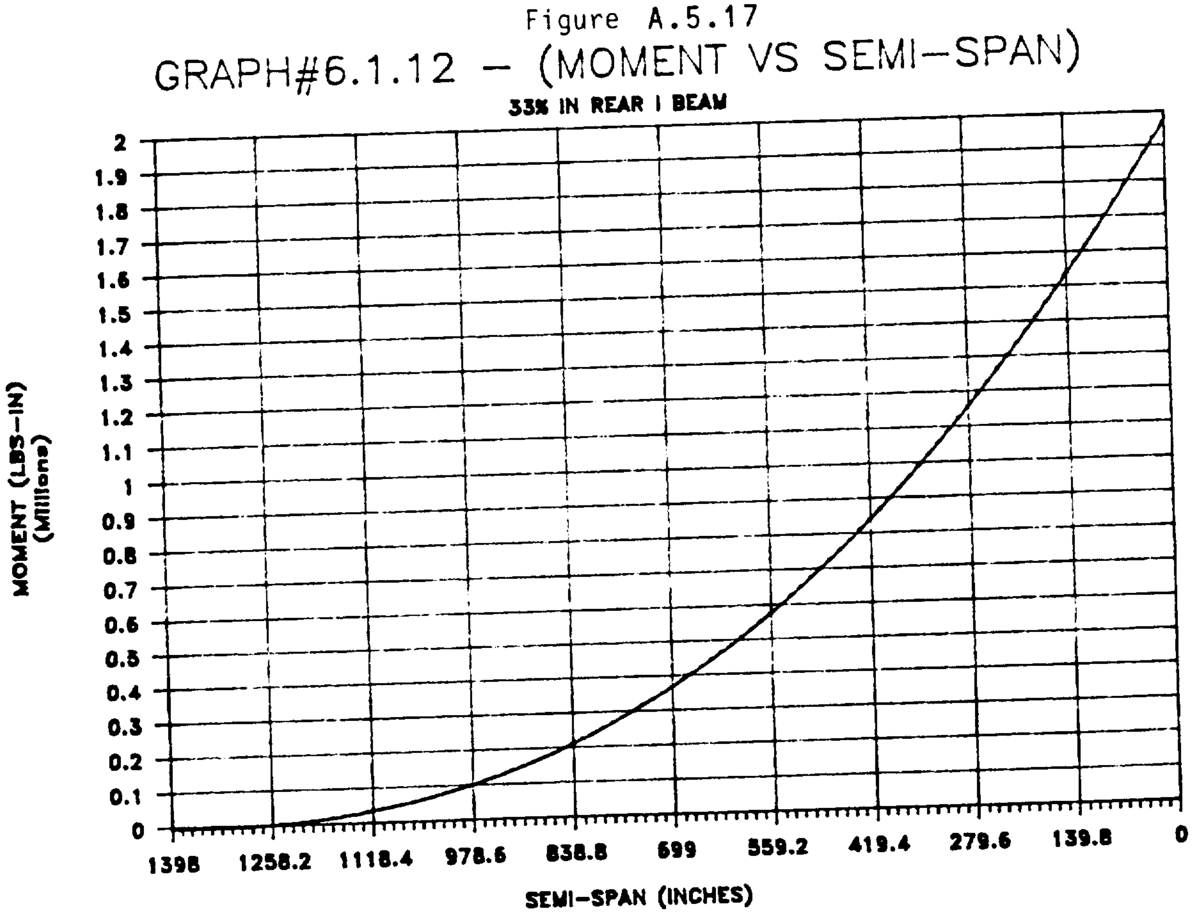


Figure A.5.18

GRAPH#6.1.13 – (LOAD VS. SEMI–SPAN)

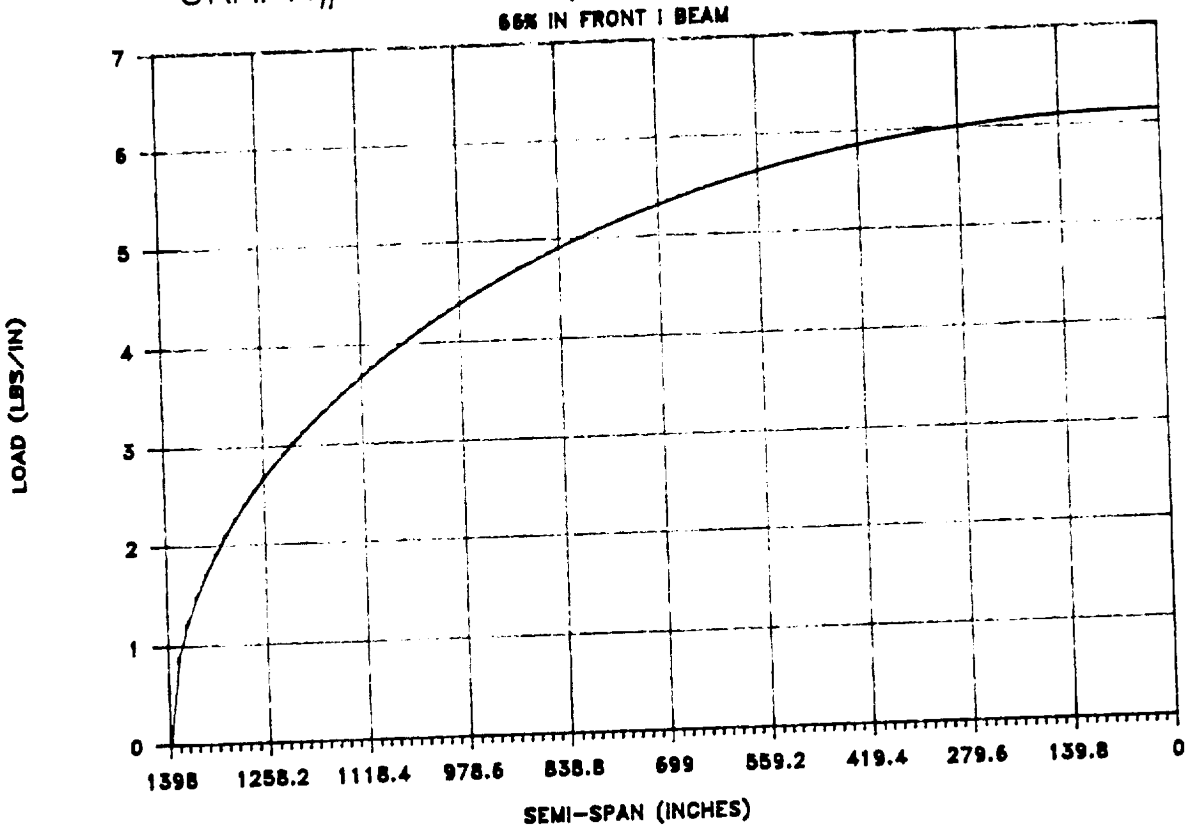


Figure A.5.19

GRAPH#6.1.14 – (SHEAR VS. SEMI–SPAN)

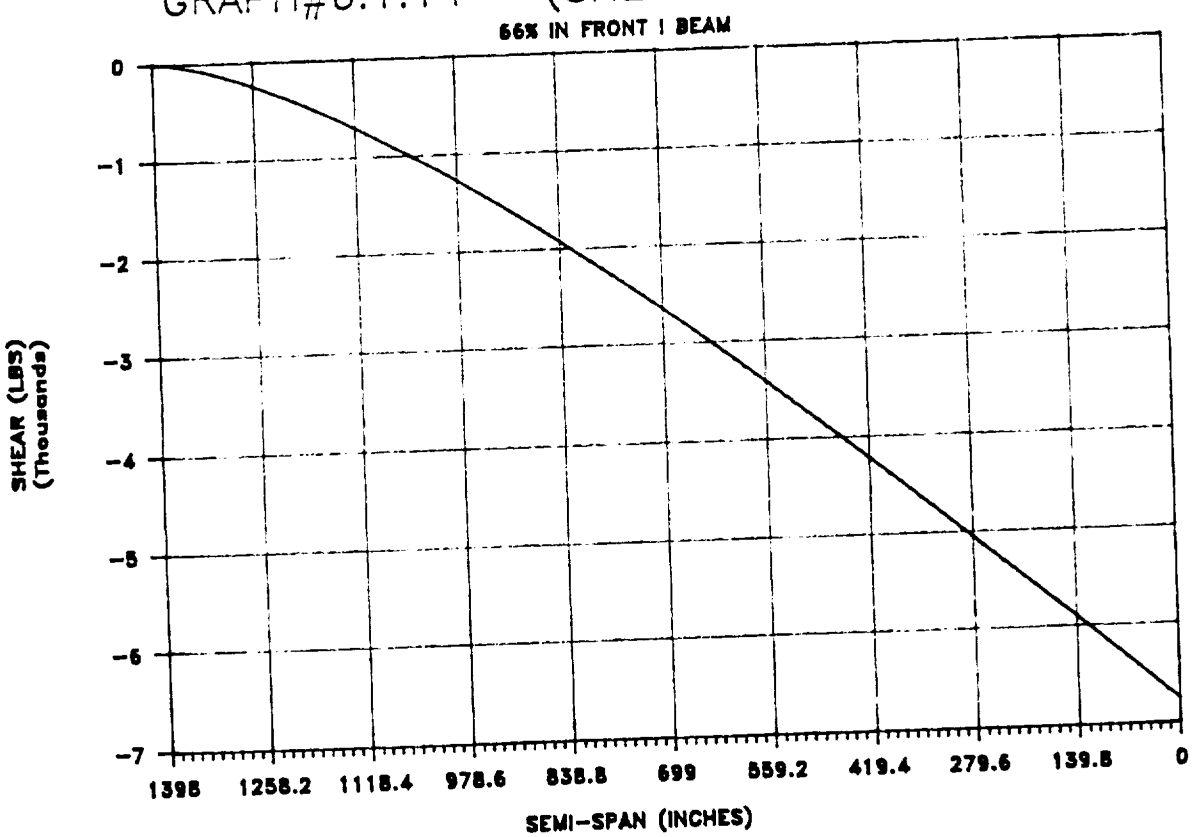


Figure A.5.20

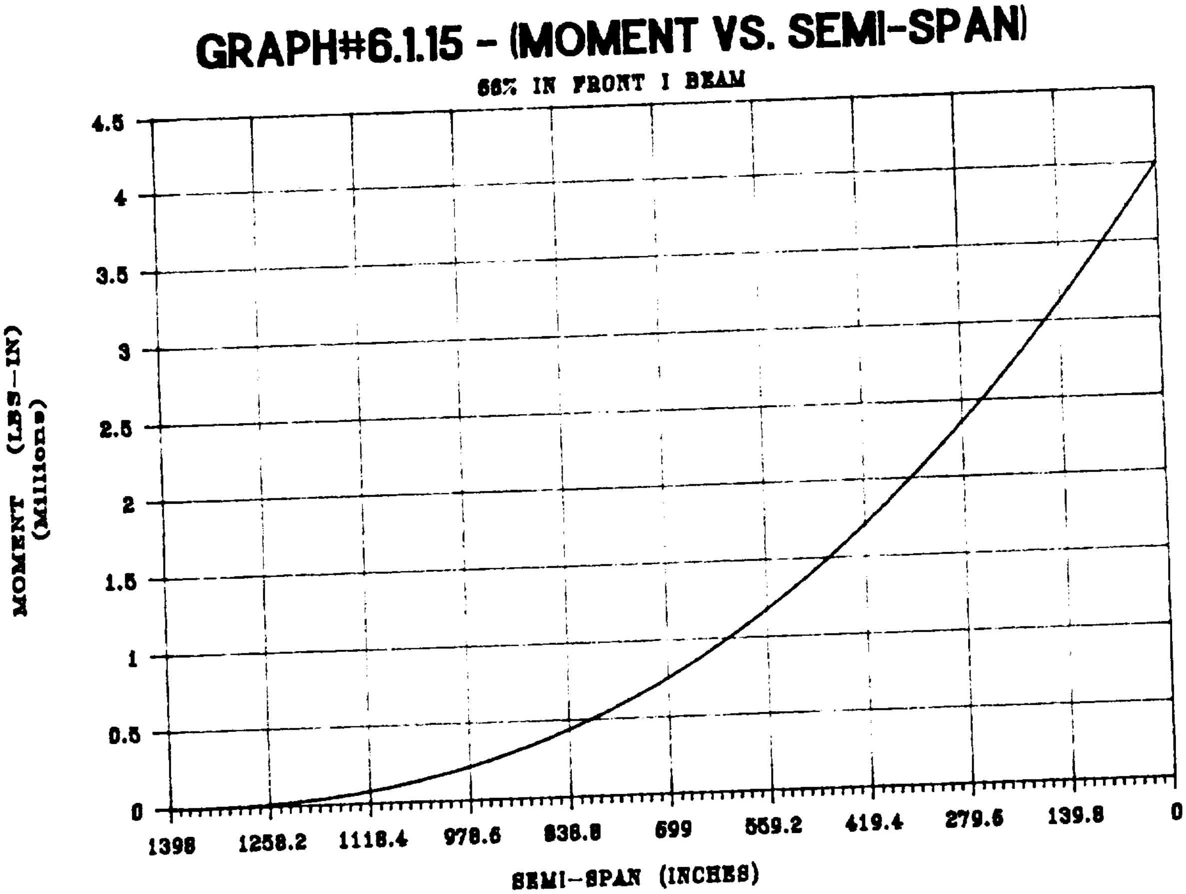

GRAPH#6.1.15 - (MOMENT VS. SEMI-SPAN)
66% IN FRONT I BEAM
MOMENT (LBS-IN)
(Millions)
4.5
4
3.5
3
2.5
2
1.5
1
0.5
0
1398
1258.2
1118.4
978.6
838.8
699
559.2
419.4
279.6
139.8
0
SEMI-SPAN (INCHES)

Figure A.5.21
Front & Rear Spar Properties

Section	Front Ave.moment I beam 66% (lb in)	Front Thickness (in)	Rear Ave. moment I beam 33% (lb in)	Rear Thickness (in)
1	3,595,672	0.1	1,771,526	0.11
2	2,768,463	0.08	1,362,724	0.087
3	2,054,949	0.061	1,012,438	0.07
4	1,460,337	0.045	719,483	0.047
5	978,358	0.04	482,020	0.04
6	603,805	0.04	297,484	0.04
7	329,770	0.04	162,472	0.04
8	147,242	0.04	72,543	0.04
9	44,366	0.04	21,858	0.04
10	4,672	0.04	2,302	0.04

Figure A.5.22

Front & Rear Spar Properties

Section	Front Area (in^2)	Front I_{xx} (in^4)	Rear Area (in^2)	Rear I_{xx} (in^4)
1	2.16	1.8E-3	1.584	1.597E-3
2	1.728	9.216E-4	1.253	7.90E-4
3	1.318	4.086E-4	1.01	4.116E-4
4	0.972	1.64E-4	0.677	1.25E-4
5	0.864	1.152E-4	0.576	7.68E-5
6	0.864	1.152E-4	0.576	7.68E-5
7	0.864	1.152E-4	0.576	7.68E-5
8	0.864	1.152E-4	0.576	7.68E-5
9	0.864	1.152E-4	0.576	7.68E-5
10	0.864	1.152E-4	0.576	7.68E-5

Figure A.5.23

Pressure distribution over the wing

Section	W(y) (lb/in)	Total Pressure (psi)	Top Pressure (psi)	Bottom Pressure (psi)
1	9.167	0.0490	0.3675	0.01225
2	9.077	0.0485	0.0364	0.0121
3	8.894	0.0476	0.0357	0.0119
4	8.613	0.04605	0.0344	0.0115
5	8.223	0.0440	0.0330	0.0110
6	7.708	0.0412	0.0309	0.0103
7	7.040	0.0376	0.0282	0.0094
8	6.170	0.0330	0.0248	0.0083
9	4.993	0.0267	0.0200	0.0068
10	3.187	0.017	0.0128	0.0042

Figure A.5.24
Pressure Calculations
for Wing

section	W(x)	span mid point	lift coef CL=0.6 L=20154
1	109.9922	5.75	2.353117
2	108.8839	17.25	2.329407
3	106.6329	28.75	2.281249
4	103.1642	40.25	2.207042
5	98.34923	51.75	2.104032
6	91.97668	63.25	1.967701
7	83.69155	74.75	1.790454
8	72.84414	86.25	1.558350
9	58.01457	97.75	1.241133
10	34.38808	109.25	0.735680

average 1.856821

lift coef	A Cp	B Cp	C Cp	D Cp	E Cp	F Cp
1.271648	2.2	0.952	-0.024	0.388	0.27	0.346
1.258835	2.153	0.946	-0.0237	0.387	0.244	0.357
1.232809	2.057	0.932	-0.0243	0.386	0.259	0.374
1.192707	1.916	0.912	-0.0252	0.383	0.247	0.397
1.137040	1.721	0.883	-0.026	0.379	0.231	0.422
1.063365	1.478	0.845	-0.0278	0.374	0.209	0.443
0.967579	1.188	0.796	-0.0303	0.368	0.181	0.446
0.842169	0.841	0.733	-0.0342	0.361	0.143	0.419
0.670721	0.44	0.646	-0.0404	0.351	0.092	0.311
0.397569	-0.036	0.509	-0.0528	0.336	0.0076	-0.0195

section	Top front A	Top Midd B	Top Rear C	Bottom Rear D	Bottom Midd E	Bottom Front F	TOT PRESS
1	0.089246	0.038620	-0.00097	0.015739	0.010953	0.014117	0.054319
2	0.087340	0.038377	-0.00096	0.015699	0.009895	0.014482	0.053277
3	0.083445	0.037809	-0.00098	0.015658	0.010506	0.015171	0.052480
4	0.077725	0.036939	-0.00102	0.015537	0.010019	0.016105	0.050595
5	0.069515	0.035821	-0.00105	0.015374	0.009370	0.017119	0.047909
6	0.059957	0.034280	-0.00112	0.015171	0.008479	0.017971	0.044372
7	0.048193	0.032292	-0.00122	0.014928	0.007342	0.018173	0.039834
8	0.034116	0.029736	-0.00138	0.014644	0.005801	0.016997	0.033861
9	0.017849	0.026207	-0.00163	0.014238	0.003732	0.012616	0.025735
10	-0.00146	0.020649	-0.00214	0.013690	0.000308	-0.00079	0.012788

in (lb/in^2)

Figure A.5.25

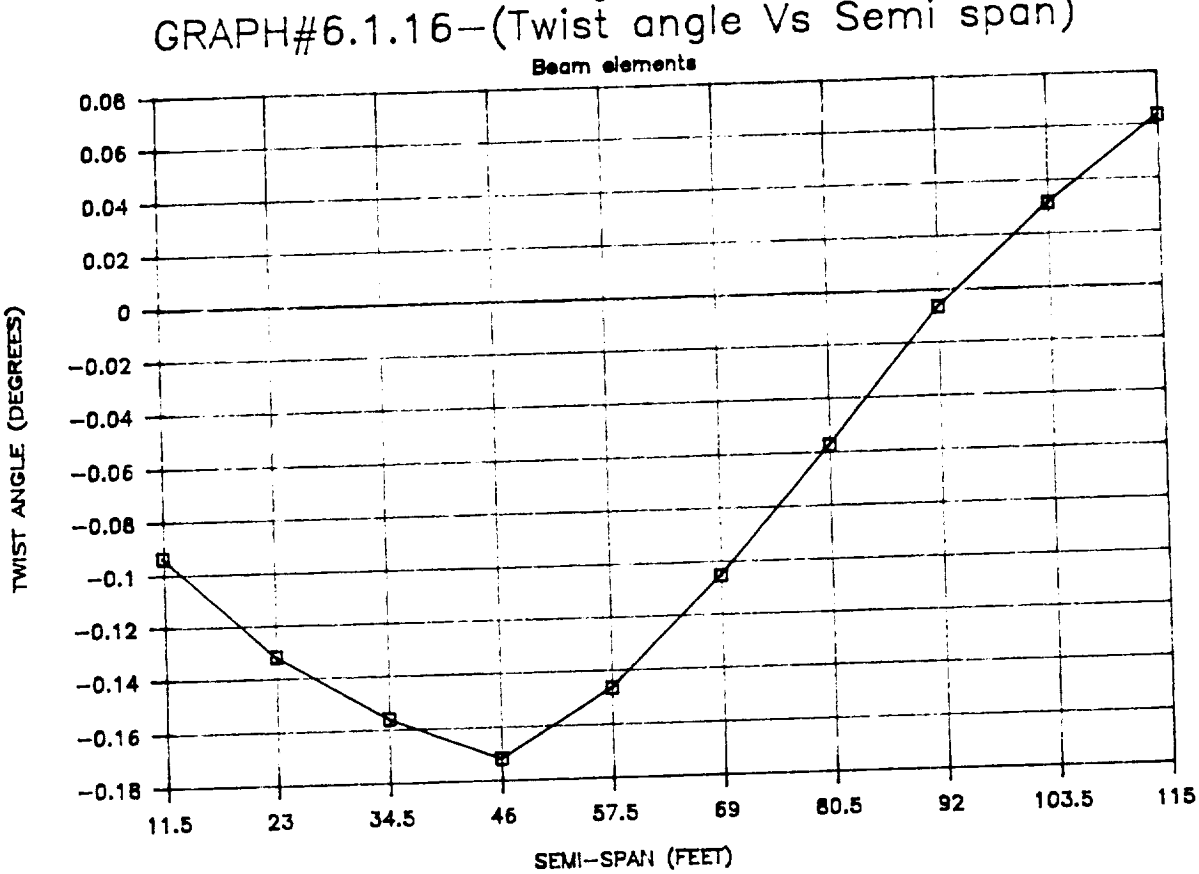
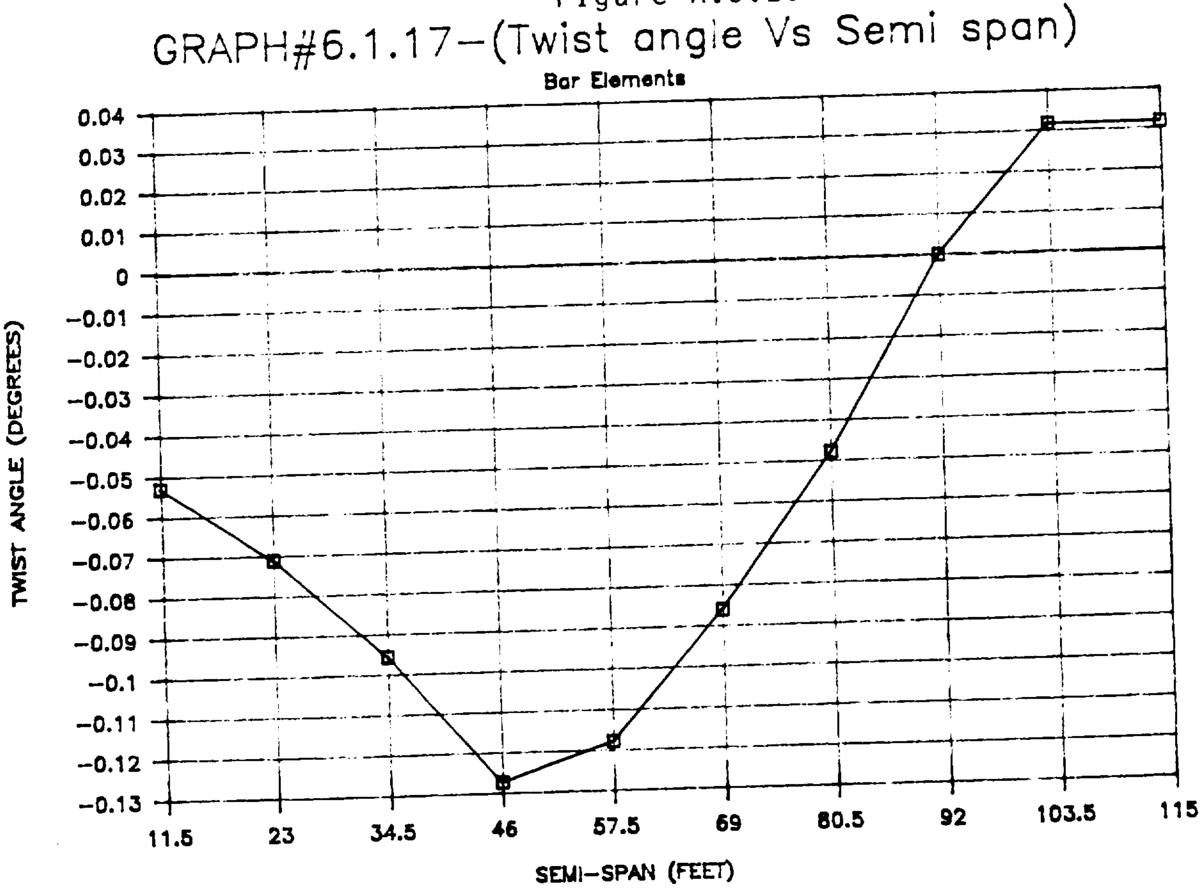
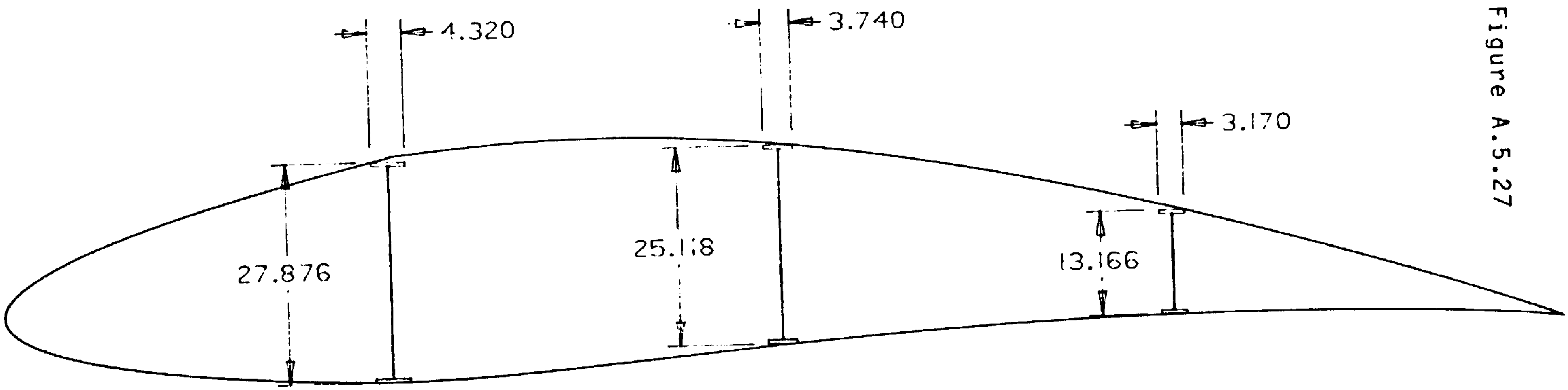


Figure A.5.26



C-2

Figure A.5.27



ALL FLANGES ARE 0.5" THICK

WPI CAD LABORATORY	TITLE: STRUCTURAL CROSS-SECTION	NO: 1
SCALE: .05 DATE: 4/8/90	DRAWN BY: TOM JUTRAS	SHEET: 1

Figure A.5.28

Force Calculations
for Wing

SECTION	TOP FRONT 1	TOP MIDD 2	TOP MIDD3	TOP REAR 4	BOTTOM REAR 5	BOTTOM MIDD 6	BOTTOM MIDD 7	BOTTOM FRONT 8	CALCULATED FORCE	EXPECTED FORCE
1	140.0870	204.7426	141.1646	43.10371	53.87963	53.87963	8.620742	-4.31037	642.1677	643.6616
2	139.7361	204.2298	140.8110	42.99574	53.74468	53.74468	8.599149	-4.29957	641.5627	642.0496
3	139.0317	203.2000	140.1012	42.77900	53.47375	53.47375	8.555801	-4.27790	639.3377	638.8131
4	137.9684	201.6461	139.0297	42.45182	53.06477	53.06477	8.490364	-4.24518	635.4708	633.9272
5	136.5377	199.5551	137.5880	42.01160	52.51451	52.51451	8.402321	-4.20116	629.9226	627.3535
6	134.7280	196.9101	135.7643	41.45477	51.81846	51.81846	8.290954	-4.14547	622.6397	619.0384
7	132.5237	193.6885	133.5431	40.77652	50.97066	50.97066	8.155305	-4.07765	613.5508	608.9103
8	129.9047	189.8608	130.9040	39.97070	49.96337	49.96337	7.994140	-3.99707	602.5641	596.8770
9	126.8455	185.3896	127.8212	39.02939	48.78673	48.78673	7.805878	-3.90293	589.5621	582.8205
10	123.3131	180.2269	124.2617	37.94251	47.42814	47.42814	7.588502	-3.79425	574.3948	566.5903
11	119.2656	174.3113	120.1831	36.69713	45.87141	45.87141	7.339426	-3.66971	556.8698	547.9933
12	114.6483	167.5632	115.5304	35.27647	44.09559	44.09559	7.055294	-3.52764	536.7375	526.7787
13	109.3896	159.8771	110.2310	33.65834	42.07292	42.07292	6.731668	-3.36583	513.6678	503.6154
14	103.3910	151.1099	104.1863	31.82262	39.76578	39.76578	6.362525	-3.18126	487.2127	475.0536
15	96.51496	141.0603	97.25739	29.69691	37.12114	37.12114	5.939382	-2.96969	456.7415	443.4599
16	88.55726	129.4298	89.23847	27.24839	34.06048	34.06048	5.449678	-2.72483	421.3198	406.8565
17	79.19253	115.7429	79.80170	24.36693	30.45866	30.45866	4.873386	-2.43669	379.4581	363.8681
18	67.84056	99.15158	68.36041	20.87401	26.09252	26.09252	4.174803	-2.08740	328.5010	311.7089
19	53.24511	77.81975	53.65469	16.38311	20.47889	20.47889	3.276622	-1.63831	262.6988	244.6467
20	31.13775	45.50903	31.37727	9.580848	11.97606	11.97606	1.916169	-0.95808	162.5151	143.0695
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Figure A.5.29
Model #2

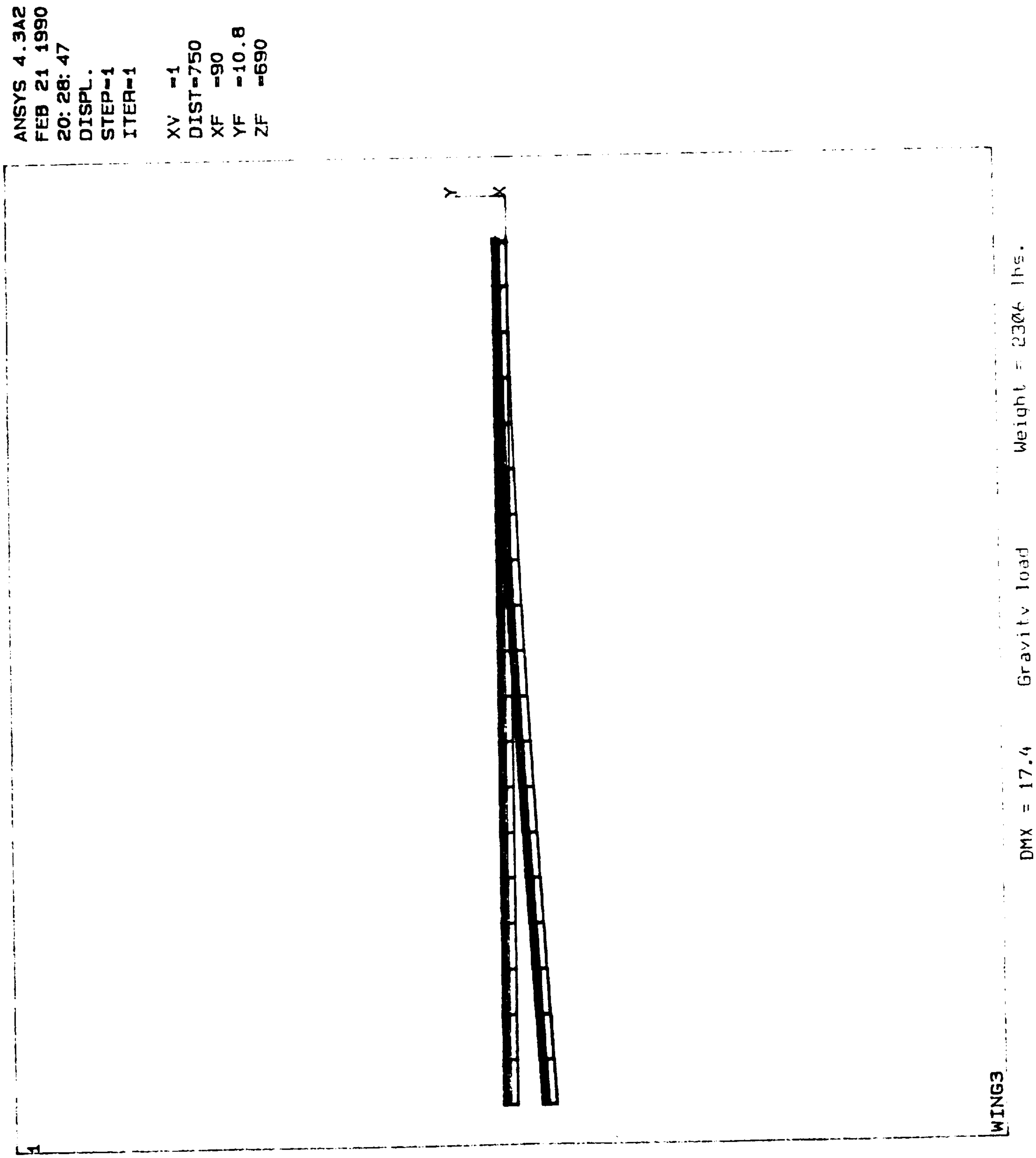

Figure A.5.30
Model #2

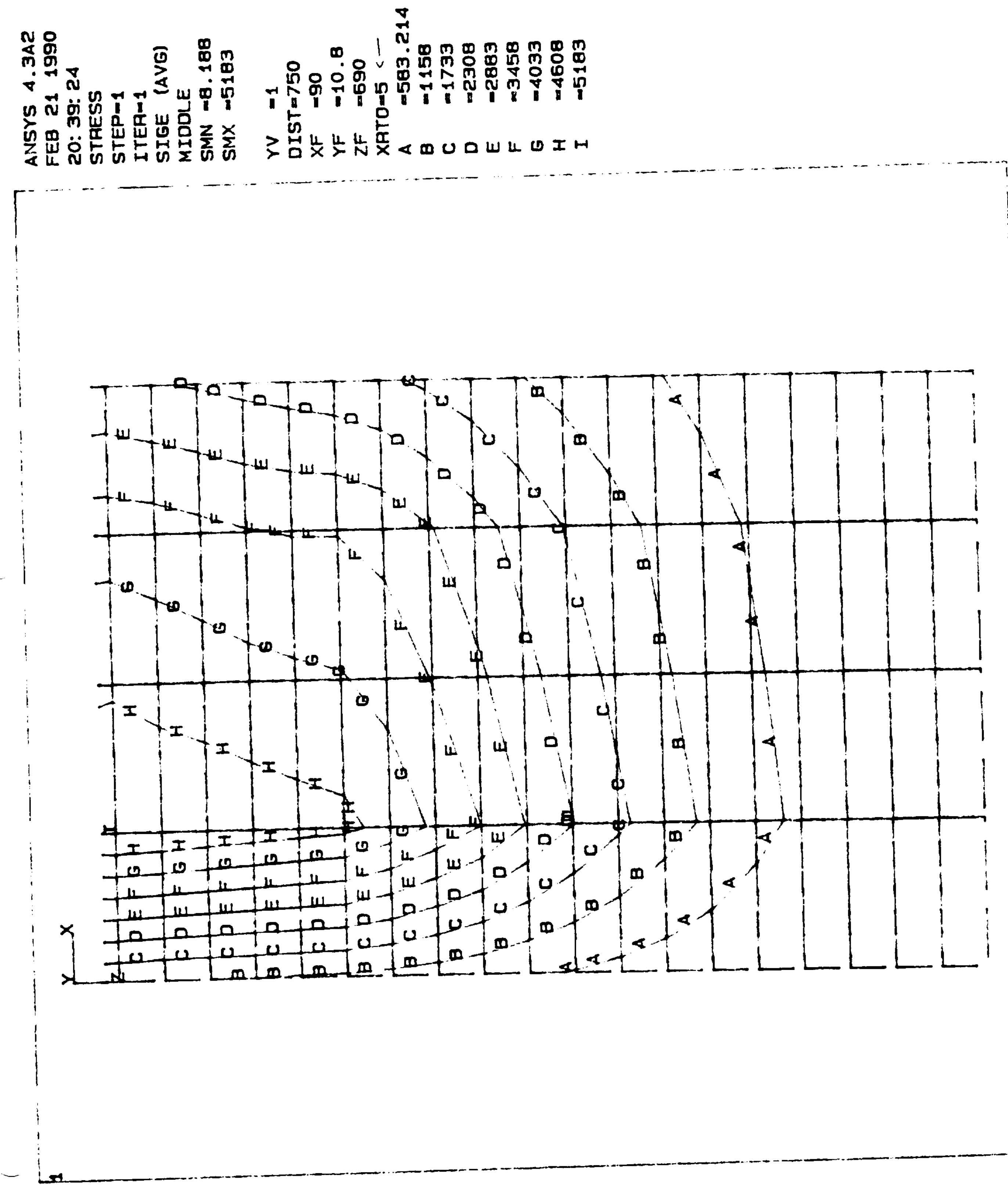

Figure A.5.31
Model #2

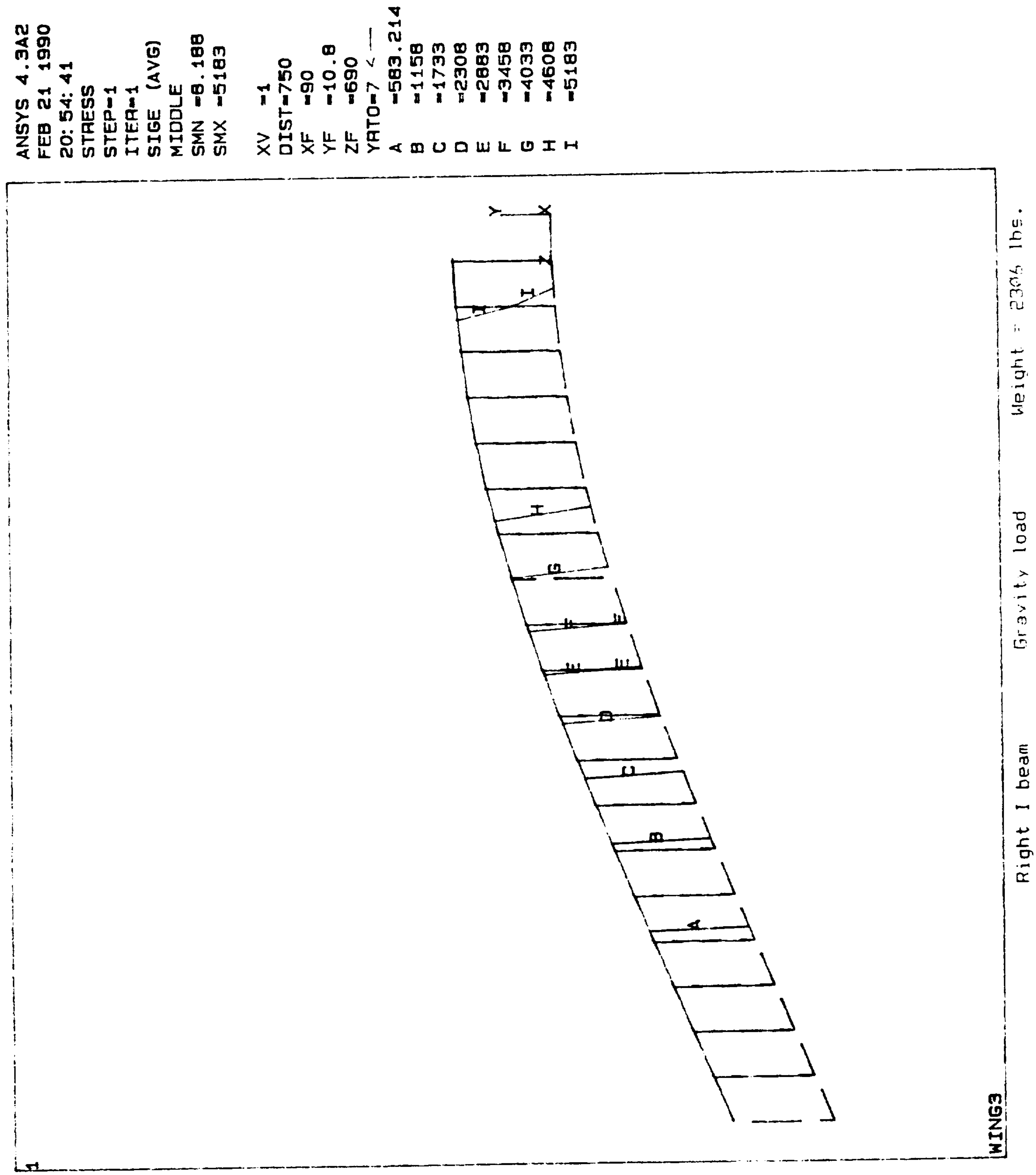

Figure A.5.32
Model #2

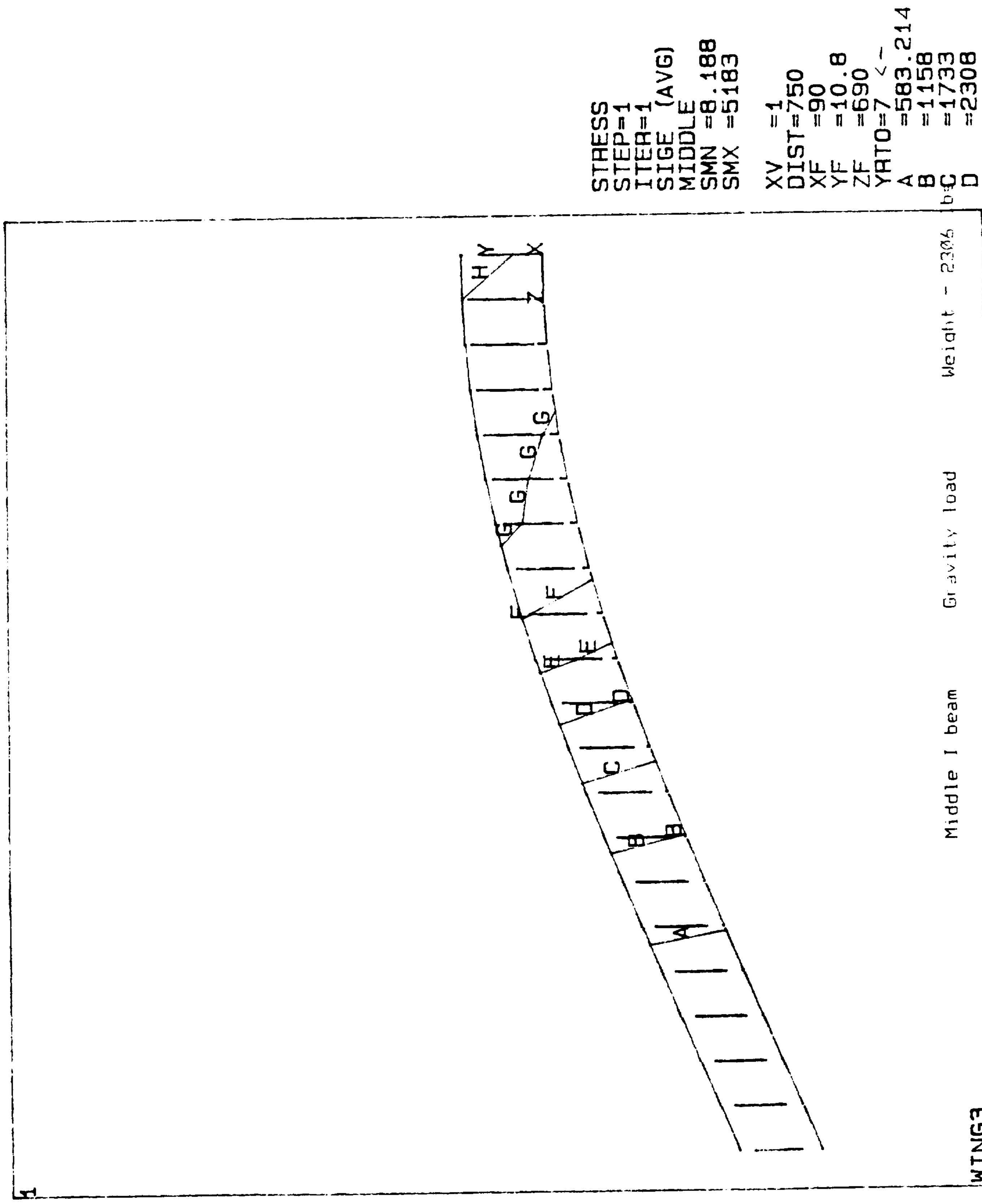

Figure A.5.33
Model #2

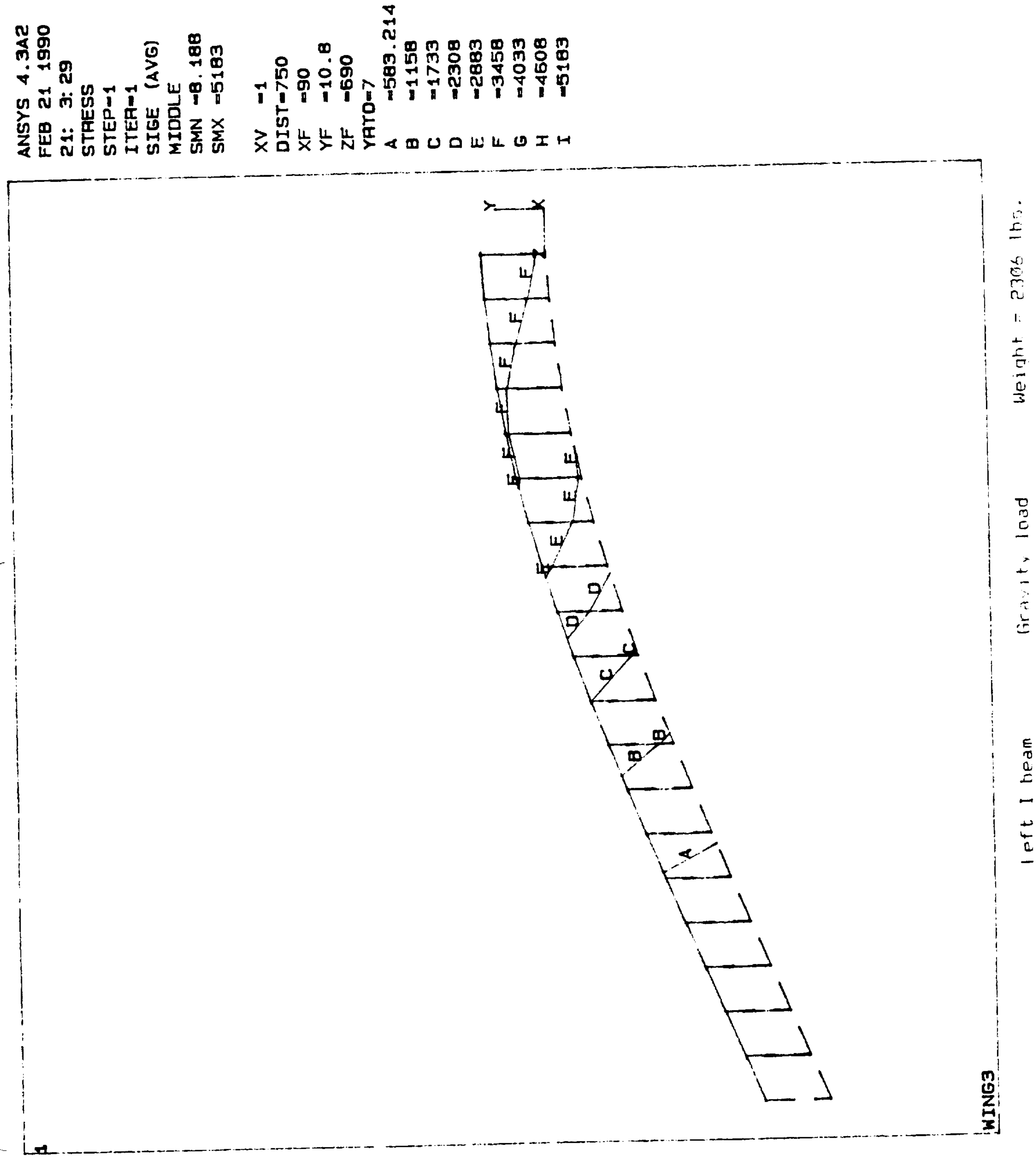

Figure A.5.34
Model #2

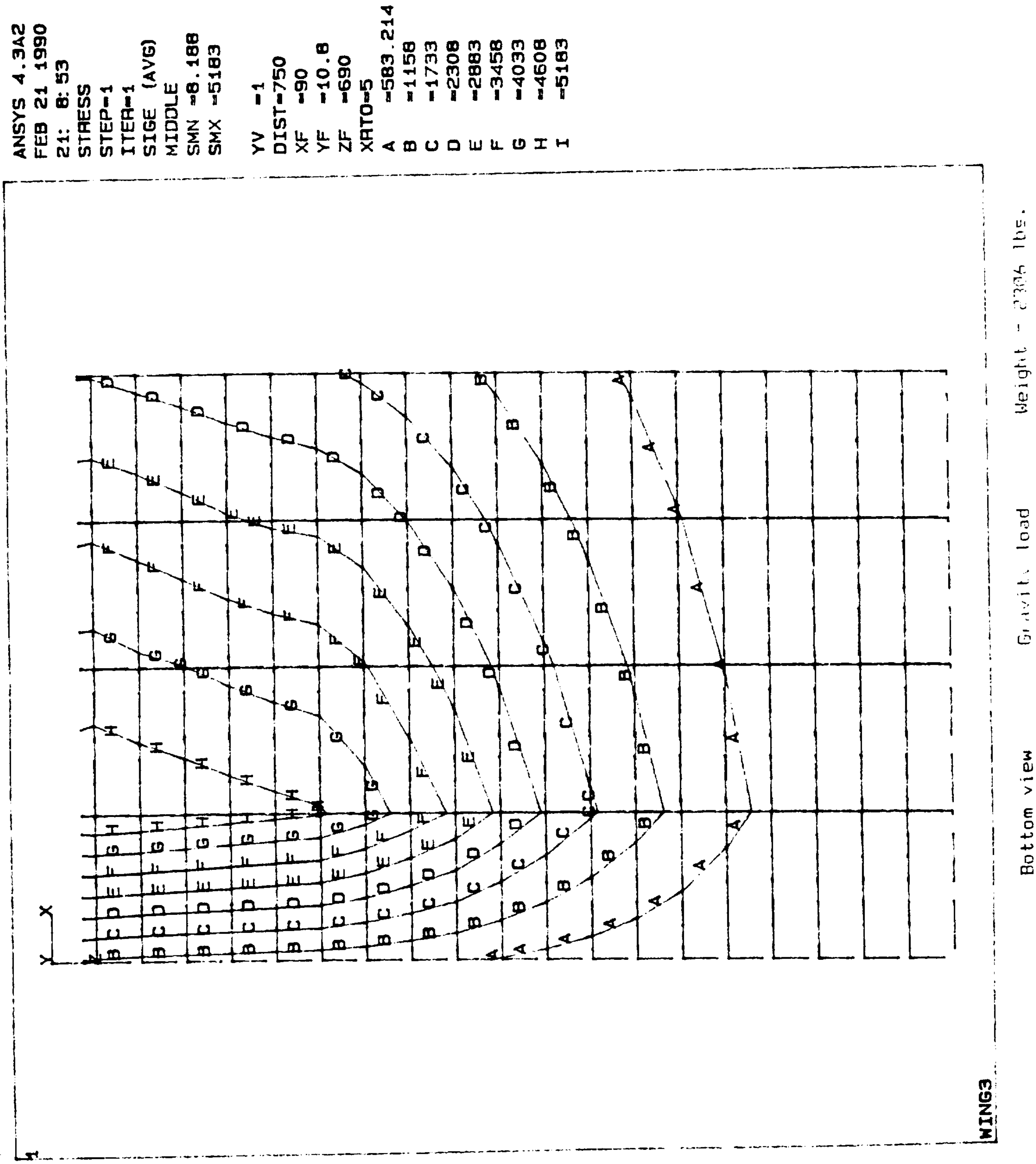

Figure A.5.35
Model #2

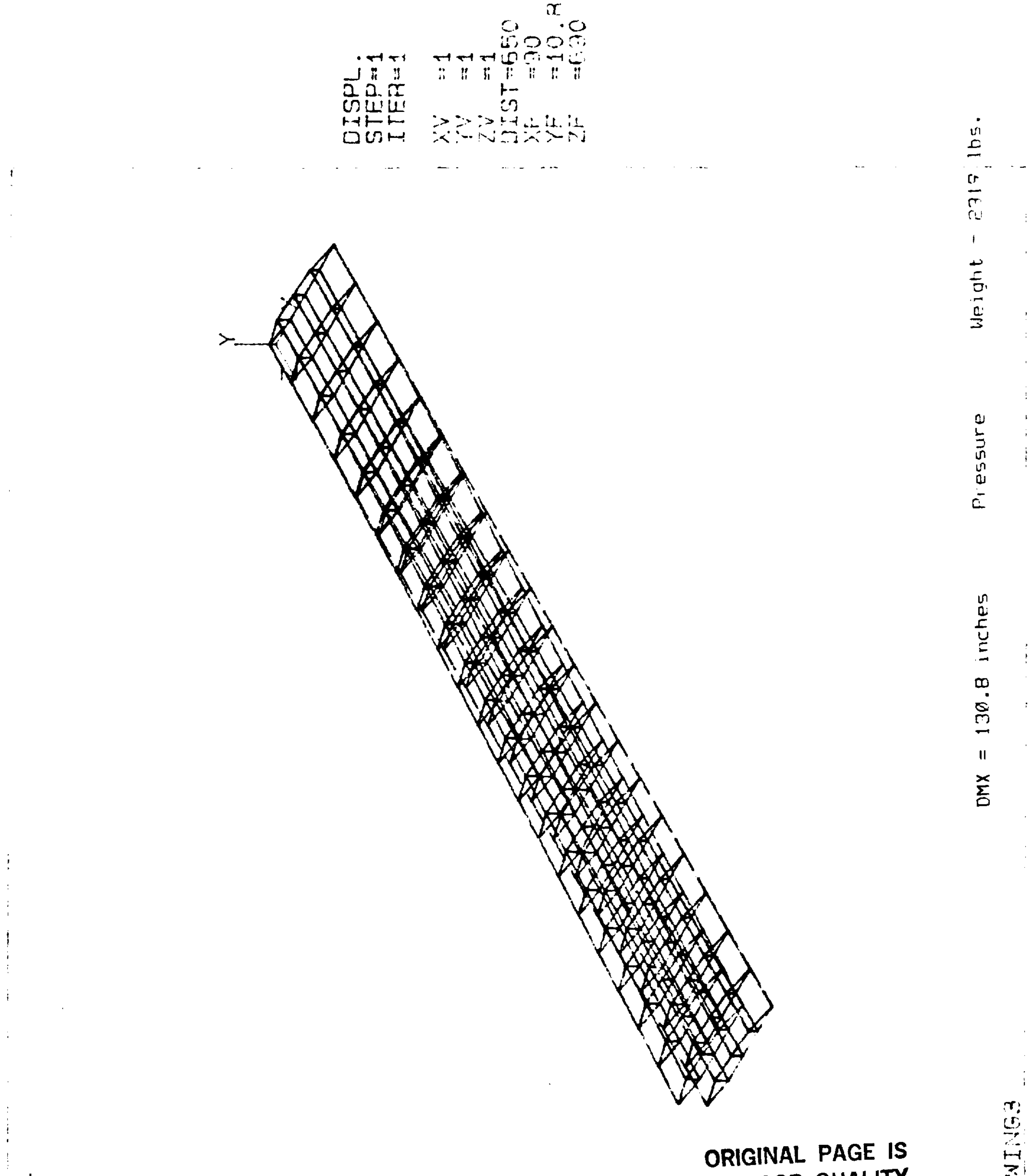

Figure A.5.36
Model #2

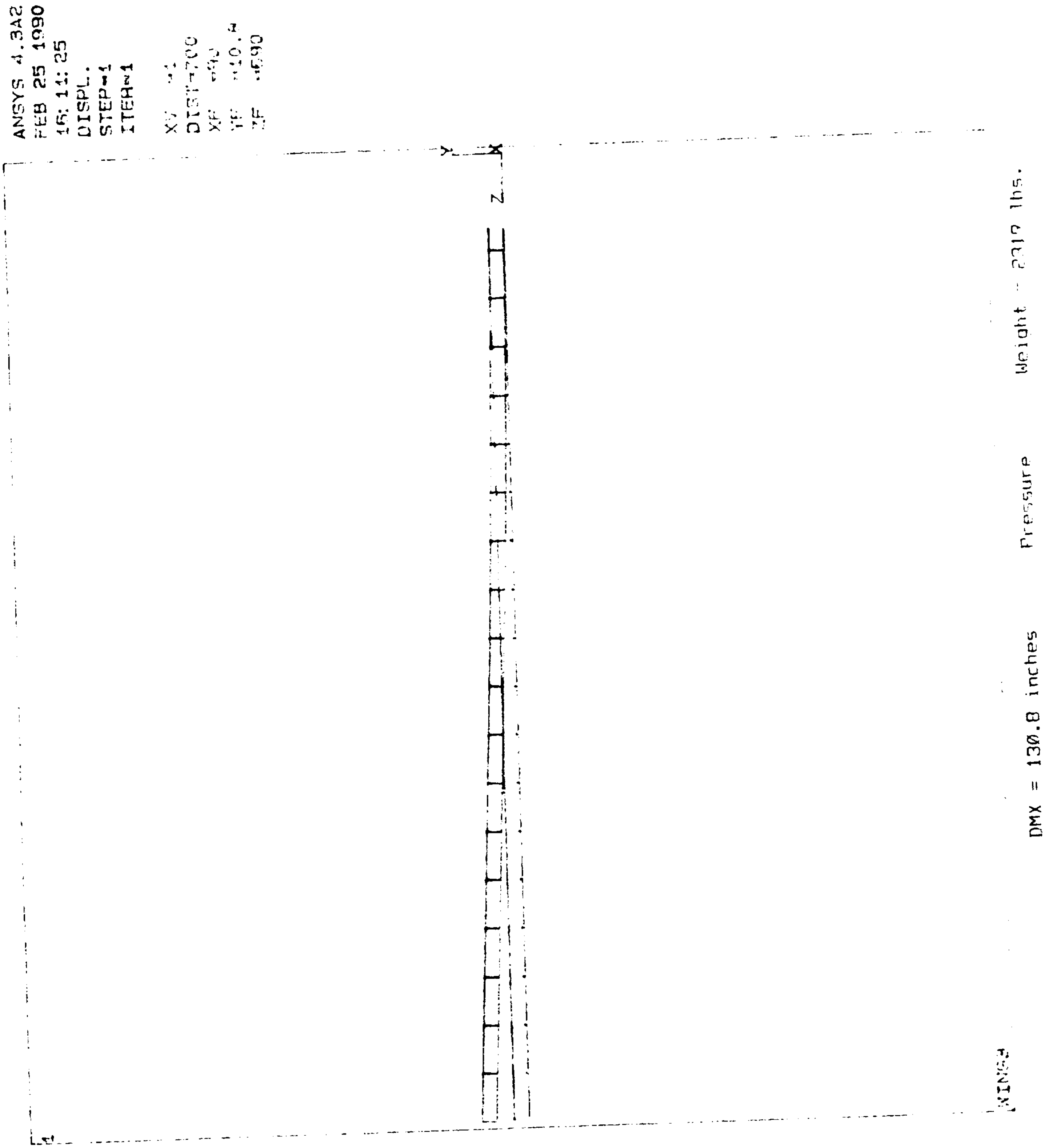

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Figure A.5.37
Model #2

Bottom view
Weight = 2319 lbs.

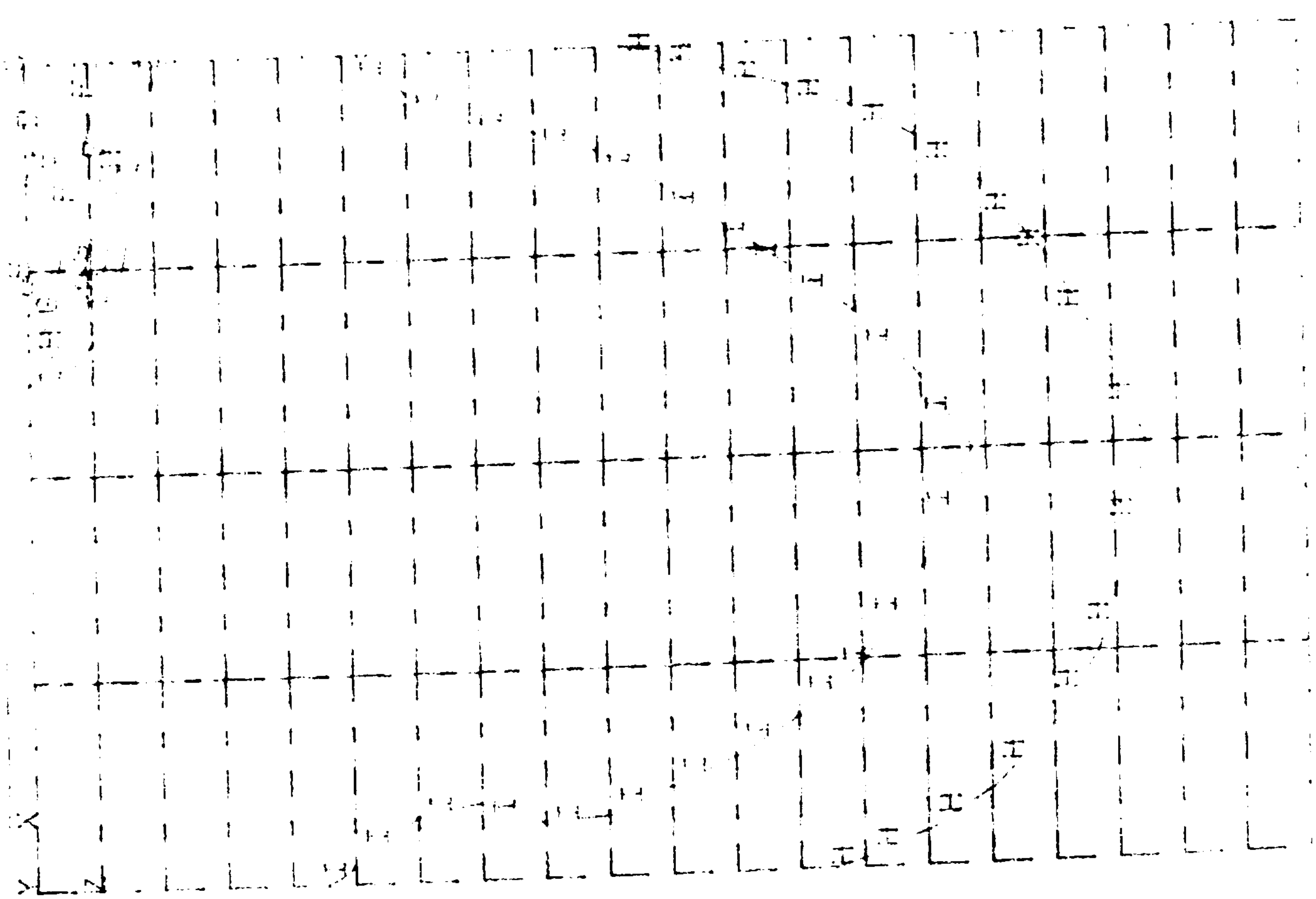

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Figure A.5.38

Model #2

Top view
Weight = 2319 lbs.

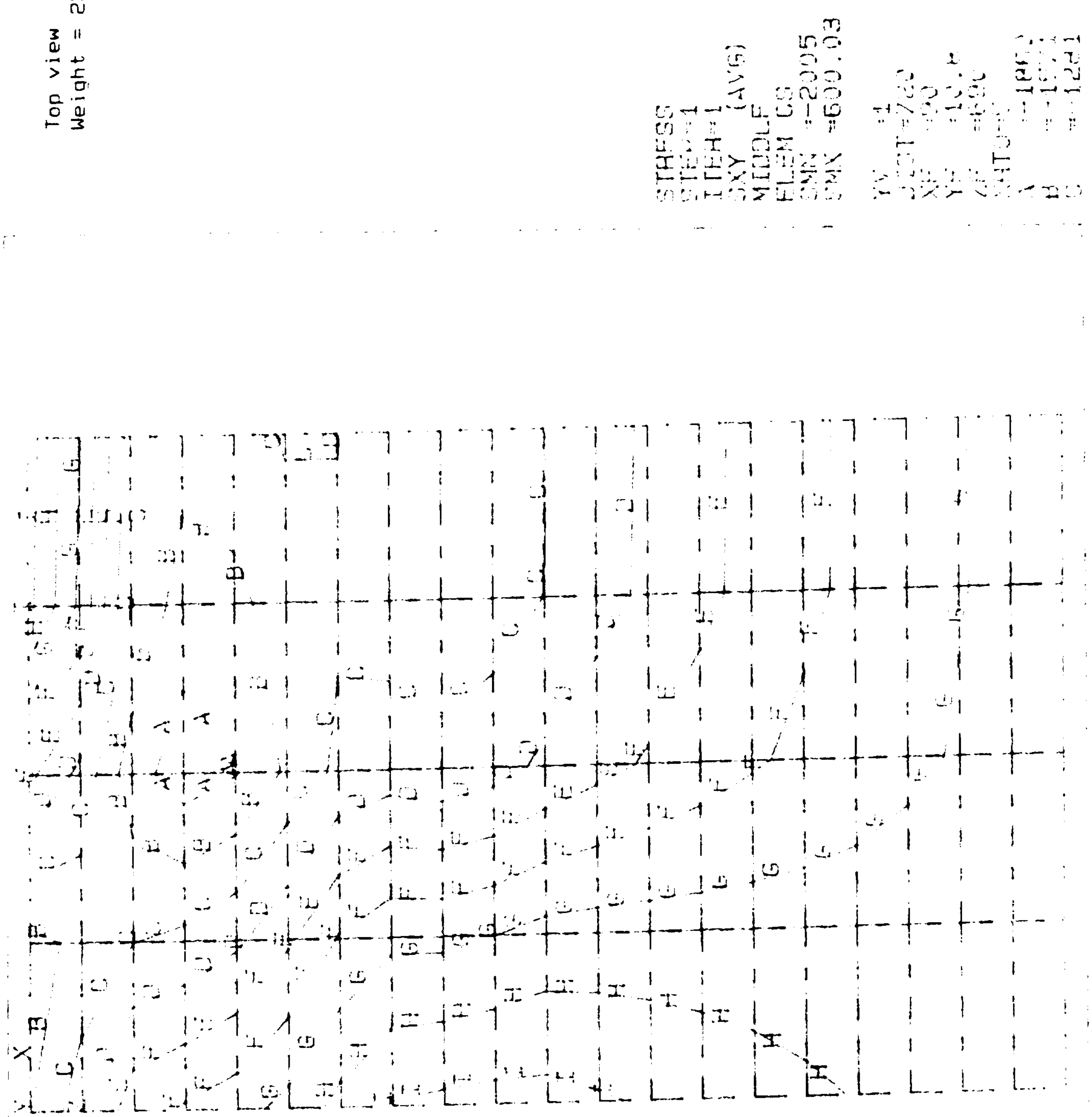

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Figure A.5.39
Model #2

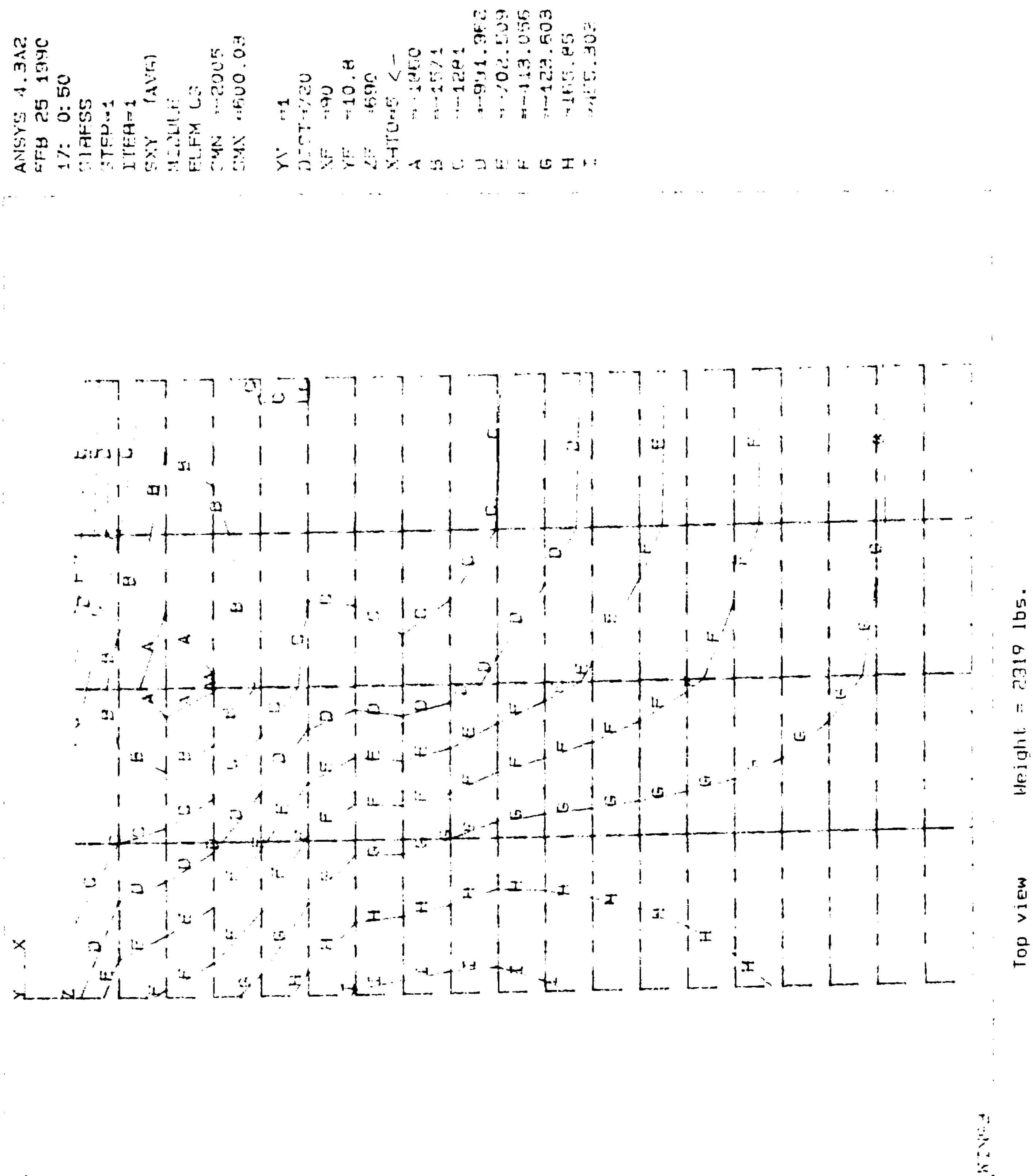


Figure A.5.40
Model #2

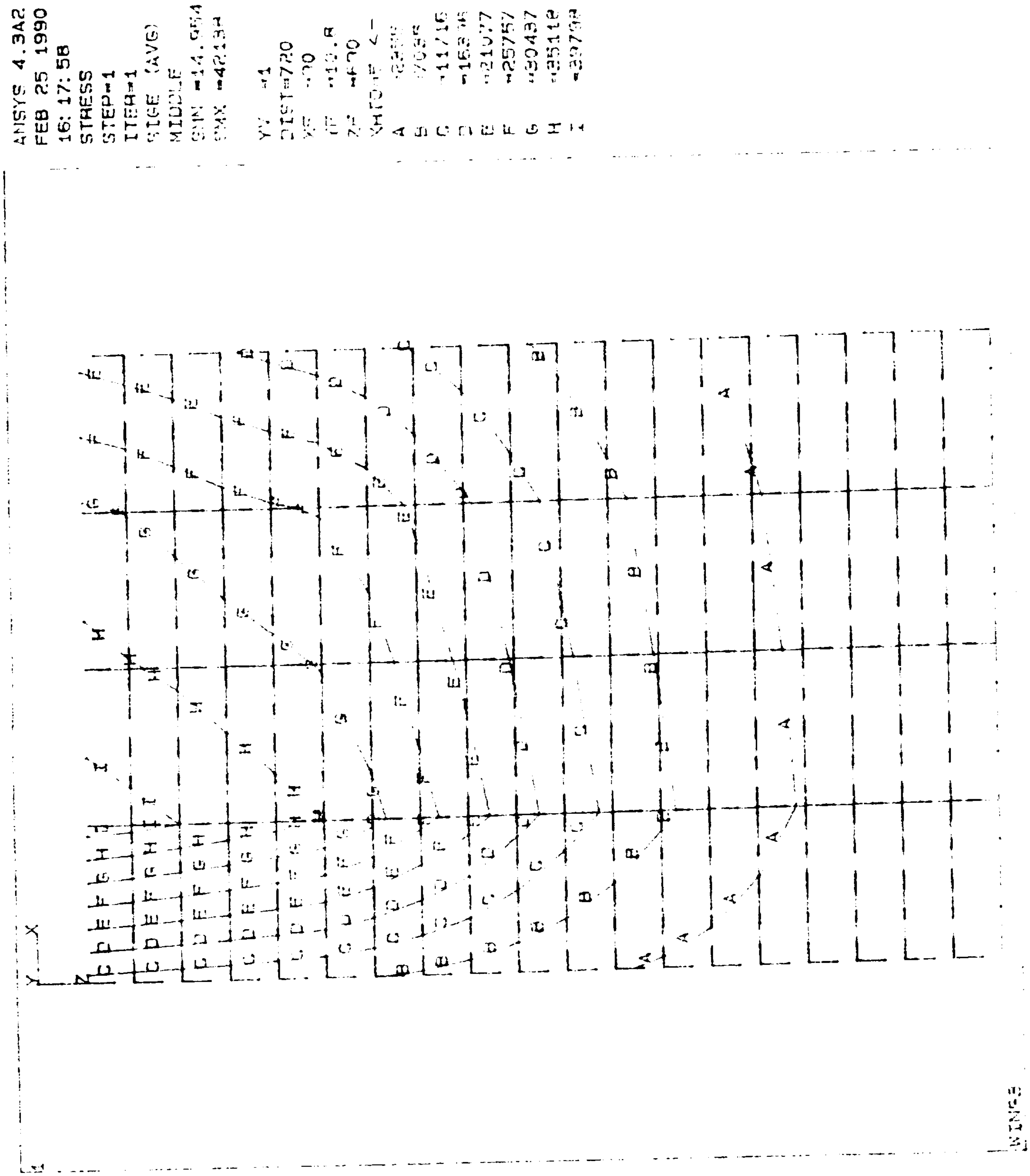


Figure A.5.41
Model #2

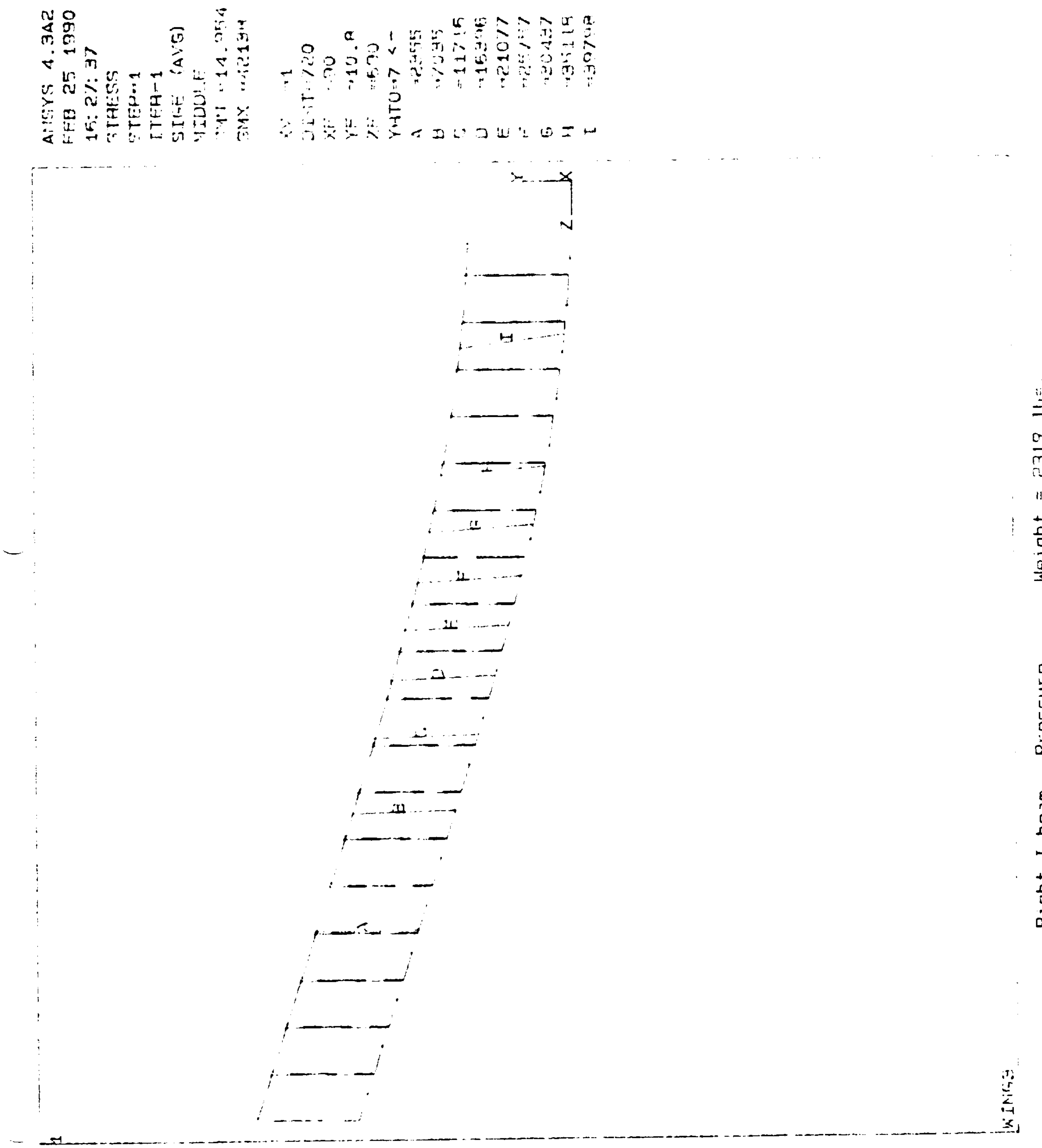
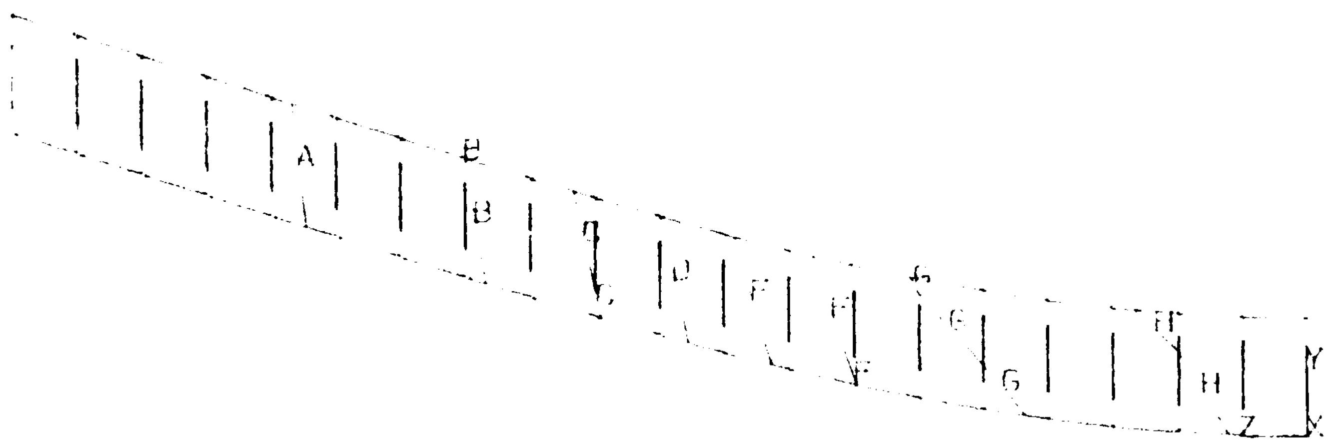


Figure A.5.42
Model #2



STRESS
STEP=1
ITER=1
SIGE (AVG)
MIDDLE
SMN =14.954
SMX =42138

XV =1
DIST=720
XF =90
YF =10.8
ZF =690
YRTO =7
A =2355
B =7035
C =11716
D =16396

Middle I beam Pressure Weight = 2319 lbs.

WING2

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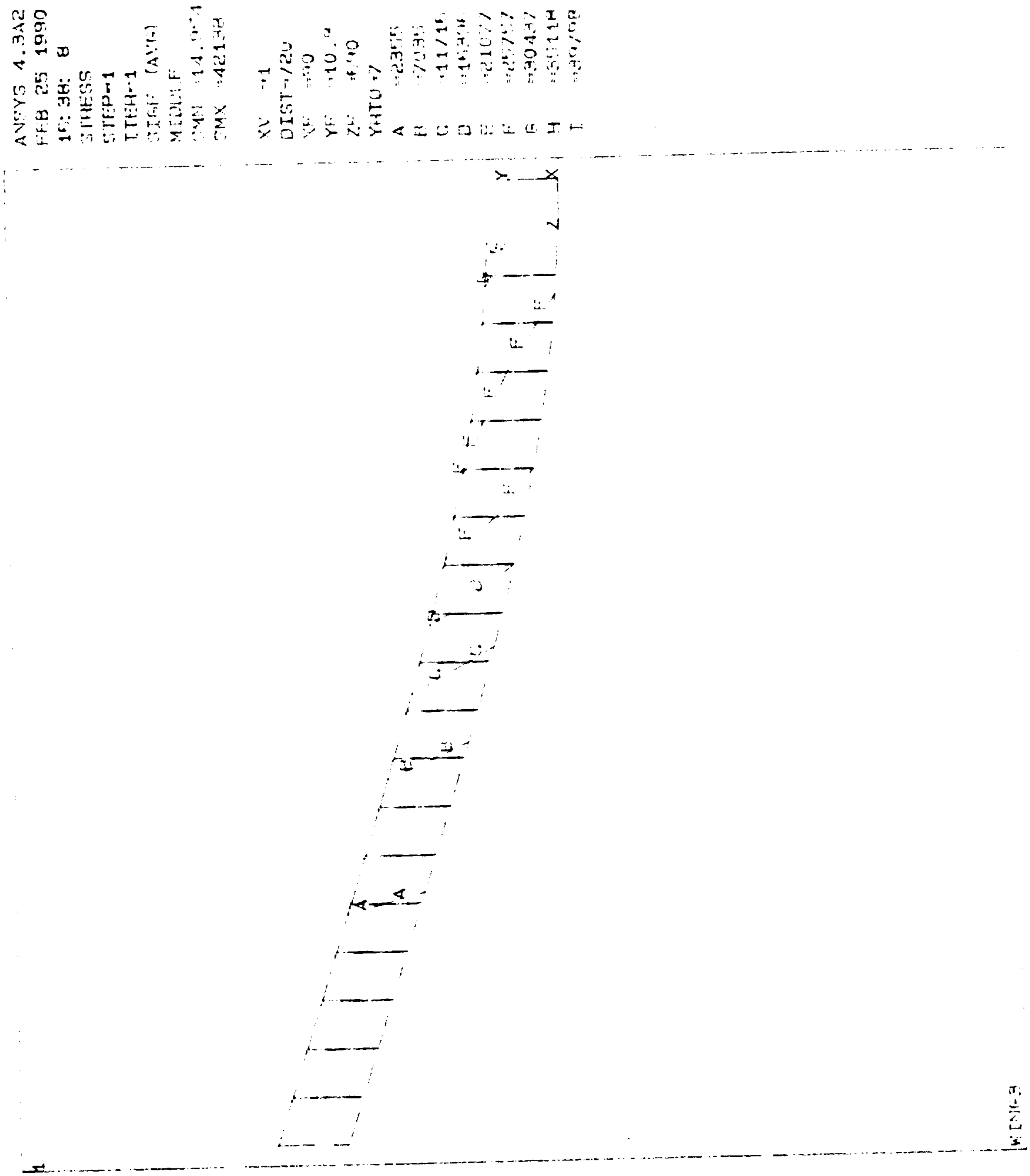
Figure A.5.43
Model #2



Left I beam Pressure Weight = 2319 lbs.

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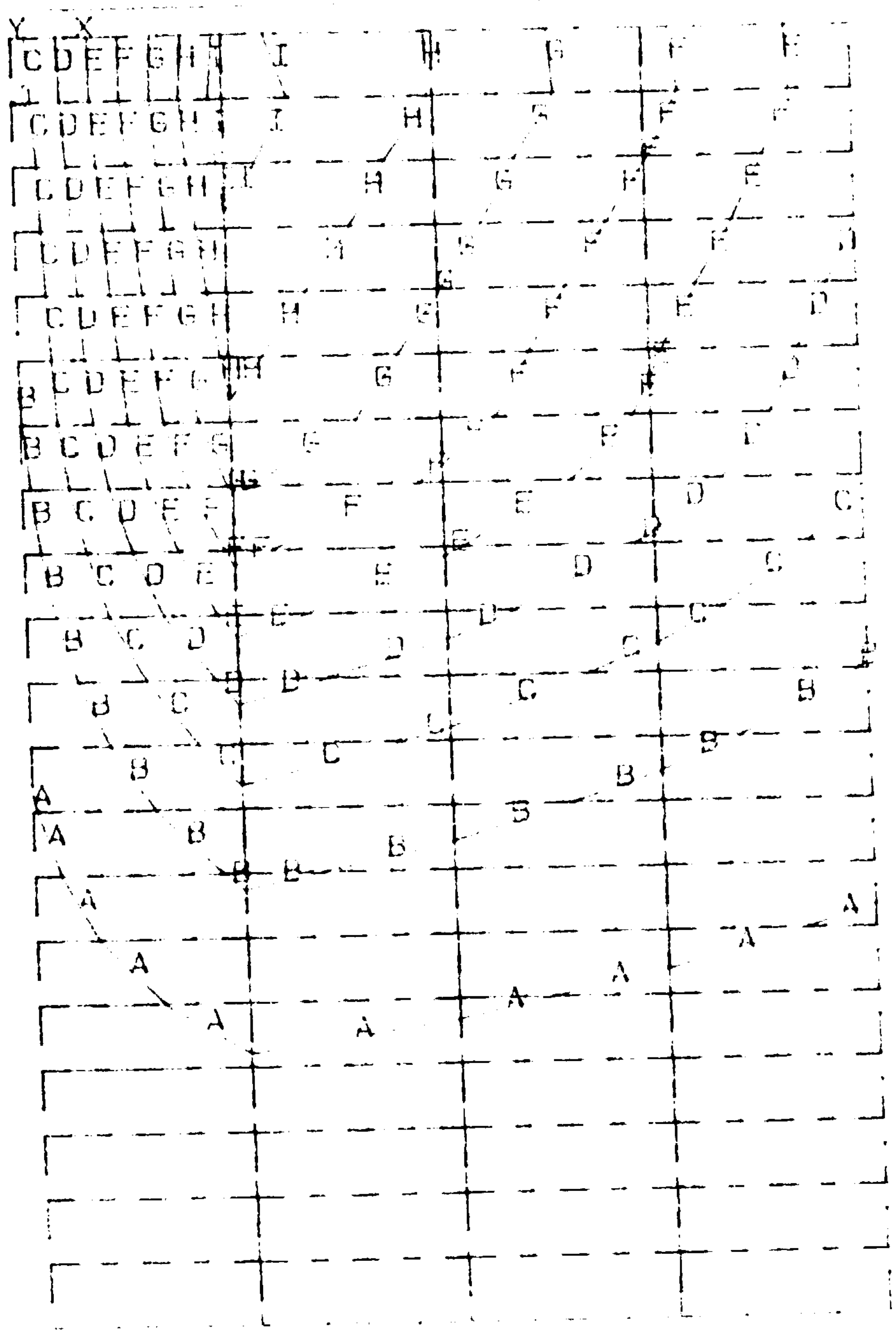
Figure A.5.44
Model #2

Bottom view
Pressure
Weight = 2319 lbs.



STRESS
STEP=1
ITER=1
SIGE (AVG)
MIDDLE
SMN =14.954
SMX =42138

YV =1
DIST=720
XF =90
YF =10.8
ZF =690
XRTO=5
A =2355
B =7055
C =11716
D =16396

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WIN53

Appendix A.6

Drive Train Sizing Calculations

The size of the propellers were calculated using the following equation from Reference 24:

$$1 - n = -\frac{1}{2} + \sqrt{(\frac{1}{4} + (T/A)-(2pV^2))} \qquad (A.6.1)$$

where:

n	-	propeller efficiency (85%)
T	-	required Thrust
A	-	actuator disc area
p	-	density of air at cruising altitude
V	-	cruising velocity

With the values entered for thrust, density and velocity, a circular cross-sectional area A of the propeller disc as described in Reference 24. The effective area for each of the propellers is half of this and the resulting propeller diameter is 10½ feet. The rotational speed of the propeller was determined using the following equation:

$$M^* = \sqrt{((V/a)^2 + (\Omega R/a)^2)} \qquad (A.6.2)$$

where:

M^*	-	maximum Mach number permitted by propeller tip = 0.8
V	-	forward velocity of aircraft
Ω	-	rotational speed of propeller
R	-	radius of propeller
a	-	speed of sound

Reference 11 described a procedure that produces an estimate of the size of a gearbox. Gearbox weight versus a factor identified as Q is plotted in Figure A.6.1. Each plot is represented by another factor called K which is selected from the table in Figure A.6.2.

$$Q = \frac{\text{Horsepower}}{\text{Pinion RPM}} * \frac{(m + 1)^3}{m} \qquad \text{(A.6.3)}$$

where:

Pinion RPM	-	equivalent to Ω
Horsepower	-	power produced by motor
m	-	gear reduction ratio

Once Q was calculated, K was selected as 600 pounds per square inch (for epicyclic spur gears in aerospace aplications). These two values were cross-referenced to produce a weight of 80 pounds which was rounded up to 100 pounds since this method of sizing gearboxes is not very accurate.

Figure A.6.1
K-Factor Table

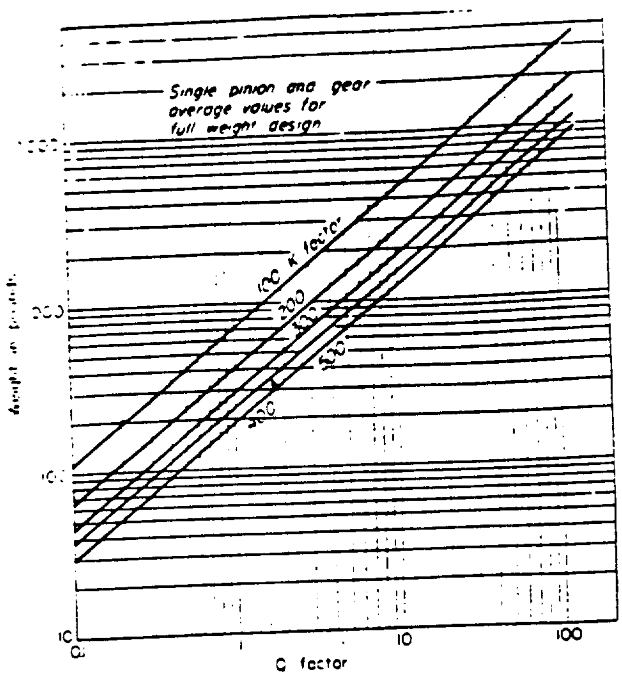
Application	Minimum hardness of steel gears		No. pinion cycles	Accuracy	K factor	
	Pinion	Gear			N/mm²	psi
Turbine driving a generator	225 HB	210 HB	10^{10}	High precision	0.69	100
	335 HB	300 HB	10^{10}	High precision	1.04	150
	59 HRC	58 HRC	10^{10}	High precision	2.76	400
Internal combustion engine driving a compressor	225 HB	210 HB	10^{9}	High precision	0.48	70
	335 HB	300 HB	10^{9}	High precision	0.76	110
	58 HRC	58 HRC	10^{9}	High precision	2.07	300
General-purpose industrial drives, helical (relatively uniform torque for both driving and driven units)	225 HB	210 HB	10^{8}	Medium high precision	1.38	200
	335 HB	300 HB	10^{8}	Medium high precision	2.07	300
	58 HRC	58 HRC	10^{8}	Medium high precision	5.52	800
Large industrial drives, spur-hoists, kilns, mills (moderate shock in driven units)	225 HB	210 HB	10^{8}	Medium precision	0.83	120
	335 HB	300 HB	10^{8}	Medium precision	1.24	180
	58 HRC	58 HRC	10^{8}	Medium precision	3.45	500
Aerospace, helical (single pair)	60 HRC	60 HRC	10^{9}	High precision	5.86	850
Aerospace, spur (epicyclic)	60 HRC	60 HRC	10^{9}	High precision	4.14	600
Vehicle transmission, helical	59 HRC	59 HRC	4×10^{7}	Medium high precision	6.20	900
Vehicle final drive, spur	59 HRC	59 HRC	4×10^{6}	Medium high precision	8.96	1300
Small commercial pitch-line speed less than 5 m/s	320 HB	Phenolic laminate	4×10^{7}	Medium precision	0.34	50
	320 HB	Nylon	10^{7}	Medium precision	0.24	35
Small gadget pitch-line speed less than 2.5 m/s	200 HB	Zinc alloy	10^{6}	Medium precision	0.10	15
	200 HB	Brass or aluminum	10^{6}	Medium precision	0.10	15

Notes: 1. The above indexes of tooth loading assume average conditions. With a special design and a favorable application, it may be possible to go higher. With an unfavorable application and/or a design that is not close to optimum, the indexes of tooth loading shown will be too high for good practice.

2. The table assumes that the controlling load must be carried for the pinion cycles shown.

Figure A.6.2

Q-Factor vs. Gearbox Weight



Gear weights plotted against Q factor for different intensities of tooth loading. (English units.)

Appendix A.7

Static Stability Derivation and Analysis Code

Contribution of Aircraft Components

It is of interest to know the contribution of the wing, fuselage, horizontal tail, and canard to the pitching moment and static stability characteristics of the airplane. Our emphasis will rely on methods that can be derived from simple theoretical considerations. These methods are generally accurate for the purposes of a preliminary design such as this. They show the relationship between the stability coefficients and the geometric and aerodynamic characteristics of the airplane. (Reference 23, p.44)

Wing Contribution

The contribution of the wing to the vehicle's static stability can be examined with the aid of Fig. A.7.1. In this sketch the wing has been replaced by its mean chord. The distances from the wing leading edge to the aerodynamic center and the CG are denoted by X_{ac} and X_{CG}, respectively. The vertical displacement of the CG is denoted by Z_{CG}. The angle the wing chord line makes with the fuselage reference line is denoted as i_w.

Summing moments about the CG, we obtain the following equation:

$$\Sigma \text{ Moments} = M_{CGw} \tag{A.7.1}$$

$$M_{CGw} = L_w \cos(\alpha_w - i_w)[X_{CG} - X_{ac}] + D_w \sin(\alpha_w - i_w)$$
$$[X_{CG} - X_{ac}] + L_w \sin(\alpha_w - i_w)[Z_{CG}]$$
$$- D_w \cos(\alpha_w - i_w)[Z_{CG}] + M_{ac_w}$$

Dividing by $\frac{1}{2}\rho V^2 Sc$ yields:

$$CM_{CGw} = C_{Lw}\ (X_{CG}/c - X_{ac}/c)\ \cos(\alpha_w - i_w) + C_{Dw} \tag{A.7.2}$$
$$(X_{CG}/c - X_{ac}/c)\ \sin(\alpha_w - i_w) + C_{Lw} Z_{CG}/c$$
$$\sin(\alpha_w - i_w) - C_{Dw} Z_{CG}/c\ \cos(\alpha_w - i_w)$$
$$+ C_{Mac_w}$$

Equation A.7.2 can be simplified by assuming that the angle of attack is small. With this assumption the following approximations can be made.

$$\cos(\alpha_w - i_w) = 1, \quad \sin(\alpha_w - i_w) = \alpha_w - i_w, \quad C_L \gg C_D$$

If we further assume that the vertical contribution is negligible, then Eq. A.7.2 reduces to:

$$C_M{}^{CGw} = C_{Mac}{}^{w} + C_{Lw}(X_{CG}/c - X_{ac}/c) \tag{A.7.3}$$

or

$$C_M{}^{CGw} = C_{Mac}{}^{w} + (C_{L0w} + dC_L/d\alpha_w * \alpha_w)\ (X_{CG}/c - X_{ac}/c) \tag{A.7.4}$$

where $C_{Lw}=C_{L0w}+dC_L/d\alpha_w \alpha_w$. Applying the condition for static stability yields:

$$C_{M0}{}^{w}=C_{Mac}{}^{w}+C_{L0w}(X_{CG}/c-X_{ac}/c) \qquad (A.7.5)$$

$$dC_M/d\alpha_w=dC_L/d\alpha_w\,(X_{CG}/c-X_{ac}/c \qquad (A.7.6)$$

Canard and Tail Contribution

The horizontal tail and canard are analyzed in much the same way as the wing. The contributions of these components may be developed with the aid of Fig. A.7.2. Both the tail and canard were designed as all moving control surfaces to increase their effectiveness and decrease overall drag. (See Figure A.7.3) The angle of attack at the tail can be expressed as $\alpha_t=\alpha_w-i_w-\epsilon+i_t$ where ϵ and i_t are the downwash and tail incidences, respectively. The angle of attack at the canard may be written $\alpha_c=\alpha_w-i_w+i_c$ where i_c is the canard incidence angle. Note that the upwash at the canard has been assumed small and neglected.

If we assume small angles and neglect the drag contribution of the tail, the total lift of the wing and tail can be expressed as:

$$L = L_w + L_t \qquad (A.7.7)$$

or

$$C_L = C_{Lw} + \eta S_t C_{Lt}/S \tag{A.7.8}$$

where

$$\eta = (\tfrac{1}{2}\rho V_t^2)/(\tfrac{1}{2}\rho V_w^2) \tag{A.7.9}$$

and η is the tail efficiency. (For this analysis, η is assumed to equal unity).

The pitching moment due to the tail can be obtained by summing the moments about the CG:

$$\begin{aligned} M_t = & -l_t[L_t\cos(\alpha FRL-\epsilon)+D_t\sin(\alpha FRL-\epsilon)] \\ & -Z_{CG}t[D_t\cos(\alpha FRL-\epsilon)-L_t\sin(\alpha FRL-\epsilon)] \\ & +M_{ac}{}^t \end{aligned} \tag{A.7.10}$$

Assuming again that $C_L \gg C_D$ and neglecting all but the first term, Eq. A.7.10 reduces to

$$M_t = -l_t L_t = -l_t C_{Lt} \tfrac{1}{2}\rho V_t^2 S_t \tag{A.7.11}$$

$$C_{Mt} = M_t/\tfrac{1}{2}\rho V^2 Sc = -l_t S_t \eta C_{Lt}/Sc \tag{A.7.12}$$

$$C_{Mt} = -V_H \eta C_{Lt} \tag{A.7.13}$$

where $V_H = l_t S_t / Sc$, the horizontal tail volume ratio. The coefficient C_{Lt} can be written as

$$C_{Lt} = (dC_L/d\alpha_t)\alpha_t = (dC_L/d\alpha_t)(\alpha_w - i_w - \epsilon + i_t) \tag{A.7.14}$$

where $dC_L/d\alpha_t$ is the slope of the tail lift curve. The downwash angle ϵ can be expressed as

$$\epsilon = \epsilon_0 + (d\epsilon/d\alpha)\alpha_w \tag{A.7.15}$$

where ϵ_0 is the downwash at zero angle of attack.

The downwash behind an elliptically loaded wing can be derived from finite wing theory:

$$\epsilon = 2C_{Lw}/\pi AR_w \tag{A.7.16}$$

where the downwash angle is in radians. The rate of change of downwash angle with angle of attack is determined by the derivative of Eq. A.7.16:

$$(d\epsilon/d\alpha) = 2(dC_L/d\alpha_w)/\pi AR_w \tag{A.7.17}$$

These expressions do not take into account the relative positions of the tail plane relative to the wing. A more accurate method would require the fabrication of a model for testing in a wind tunnel. Rewriting the tail

contribution to the pitching moment yields

$$C_{M}^{CGt}=-V_{H}\eta C_{Lt} \tag{A.7.18}$$

$$C_{M}^{CGt}=\eta V_{H}(dC_{L}/d\alpha_{t})(\epsilon_{0}+i_{w}-i_{t}) -\eta V_{H}(dC_{L}/d\alpha_{t})\alpha\ (1-d\epsilon/d\alpha) \tag{A.7.19}$$

Finally, the expressions for the intercept and slope may be written:

$$C_{MO}^{t}=\eta V_{H}(dC_{L}/d\alpha_{t})(\epsilon_{0}+i_{w}-i_{t}) \tag{A.7.20}$$

$$dC_{M}/d\alpha_{t}= -\eta V_{H}(dC_{L}/d\alpha_{t})(1-d\epsilon/d\alpha) \tag{A.7.21}$$

By similar analysis, the intercept and slope of the canard are found to be:

$$C_{MO}^{c}=\eta V_{H}(dC_{L}/d\alpha_{c})(i_{w}-i_{c}) \tag{A.7.22}$$

$$dC_{M}/d\alpha_{c}=\eta V_{H}(dC_{L}/d\alpha_{c}) \tag{A.7.23}$$

H.1.3: Fuselage Contribution

The fuselage contribution to the pitching moment curves is outlined in Ref. 23, pp. 49-51. The fuselage is divided into segments and the local induced angle due to the wing upwash or downwash for each segment can be

estimated. A computer program was written to estimate the intercept and slope of the fuselage contribution to the pitching moment curves; this code is included in this appendix.

Stick Fixed Neutral Point

The total pitching moment for the airplane can now be obtained by summing the wing, fuselage, and tail contributions.

$$C_M{}^{CG}=C_{MO}+(dC_M/d\alpha)\alpha \qquad (A.7.24)$$

where

$$C_{MO}=C_M{}^{0w}+C_M{}^{0f}+\eta V_{Ht}(dC_L/d\alpha_t)(\epsilon_0+i_w-i_t)$$
$$+\eta V_{Hc}(dC_L/d\alpha_c)(i_w-i_c) \qquad (A.7.25)$$

$$dC_M/d\alpha=(dC_L/d\alpha_w)(X_{CG}/c-X_{ac}/c)+(dC_M/d\alpha_f)$$
$$-\eta V_{Ht}(dC_L/d\alpha_t)(1-d\epsilon/d\alpha)+\eta V_{Hc}(dC_L/d\alpha_c) \qquad (A.7.26)$$

Setting $dC_M/d\alpha$ equal to zero and solving for the center of gravity position yields

$$(X_{NP}/c)=(X_{ac}/c)-((dC_M/d\alpha_f)/(dC_L/d\alpha_w))+\eta V_H$$
$$((dC_L/d\alpha_t)/(dC_L/d\alpha_w))(1-d\epsilon/d\alpha)-$$
$$\eta V_H((dC_L/d\alpha_c)/(dC_L/d\alpha_w)) \qquad (A.7.27)$$

This location, the stick fixed neutral point, is the point of neutral stability; movement of the CG behind this point will cause the airplane to become unstable.

Static Stability Analysis Code

The stability analysis of the aircraft required the creation of a computer program to quickly solve the longitudinal stability equations. This would allow us to see the effects of changes of variable values on the static stability of the aircraft, and thus to choose the best control surface configuration for the given flight conditions.

The static stability analysis program performs four interrelated functions. The first is the calculation of the moment coefficients due to the wing, tail, and canard, as well as calculating the slope and intercept of Cm vs. α for each component. The second function is the optional calculation of the Cm, and slope and intercept of Cm vs. α for the fuselage of the aircraft. The third function is the tabulation (formatted for Lotus) and output of the results, for the components and the composite aircraft, vs. α. Finally, the program calculates the neutral point of the aircraft for subsequent iterations.

All necessary geometric and performance constants are valued within the program. The distances to the center

of gravity and aerodynamic center (measured from the quarter chord of the wing), and tail and canard quarter chords (measured from the leading edge of the wing), are left as variable inputs. Also left as variable inputs are the angles of incidence of the wing, tail, and canard.

```
1 REM
2 REM ***********************************************************
3 REM **   S t a t i c   S t a b i l i t y   A n a l y s i s   **
4 REM **                  P r o g r a m                         **
5 REM **                                                        **
6 REM **        written by Noah P. Forden and Ethan Odin        **
7 REM ***********************************************************
8 REM
9 REM
20 PRINT "This version, STAT7.BAS, accepts degrees as input."
30 PRINT
100 REM          Static stability analysis program
110 REM
120 REM
130 REM  Geometric constants
140 REM
150 LET C =15.36 : REM  wing mean aerodynamic chord : ?"c =";c
160 LET SW = 3594.25 : REM  wing planform area : ?"Sw =";sw
170 LET ARW = 15.23 : REM  wing aspect ratio : ?"ARw = ";arw
180 LET ST = 520 : REM  tail planform area : ?"St =";st
190 LET ART = 5.2 : REM  tail aspect ratio : ?"ARt =";art
200 LET SC = 470 : REM  canard planform area : ?"Sc =";sc
210 LET ARC = 4.7 : REM  canard aspect ratio : ?"ARc =";arc
220 LET IFUS = 0   : REM  fuselage incidence angle : ?"If =";ifus
230 LET LT = 36.5: LET LC = 23.66: LET XAC=3.84: LET XCG=3.049
240 REM
250 REM
260 REM  Performance constants
270 REM
280 LET ETA = 1 : REM  wing/tail effective velocity ratio : ?"ETA =";eta
290 LET CLOW = .302 : REM  CL at 0 AOA : ?"CL0w =";cl0w
300 LET CMACW = -.0596 : REM  CM of the ac on the wing : ?"CMacw =";cmacw
310 LET CLAA = 2*3.14 : REM  lift-curve slope for infinite wing : ?"Claa =";claa
320 CLAW = CLAA/(1+(CLAA/(3.14159*ARW)))
330 CLAT = CLAA/(1+(CLAA/(3.14159*ART)))
340 CLAC = CLAA/(1+(CLAA/(3.14159*ARC)))
350 LET DK = .86 : REM  k2-k1 : ?"dk =";dk
360 LET A0W = -1.5 : REM AOA of the wing at 0 lift : ?"A0w =";a0w
370 REM
380 REM
390 REM
400 REM Semivariable inputs
410 REM
420 INPUT "It :";IT:IT=IT*1.745329E-02
430 INPUT "Ic :";IC:IC=IC*1.745329E-02
440 INPUT "Iw :";IW:IW=IW*1.745329E-02
450 REM
460 REM
470 REM
480 REM
490 REM Preliminary calculations
500 REM
510 VHT = (LT*ST)/(SW*C)
520 VHC = -(LC*SC)/(SW*C)
530 DEDA = (2*CLAW)/(3.14159*ARW)
540 E0 = (2*CLOW)/(3.14159*ARW)
550 REM
560 REM
570 GOTO 760
580 REM Find CMaf and CM0f
```

```
590 REM
600 INPUT "Do you need to calculate CMf ";Q$
610 IF Q$="Y" THEN GOTO 650
620 INPUT "CM0f:";CM0F
630 INPUT "CMaf:";CMAF
640 GOTO 770
650 INPUT "How many sections?";SECTIONS
660 CMAF=0 : CM0F=0
670 FOR I = 1 TO SECTIONS
680 PRINT "Length of section";I:INPUT DXF
690 INPUT "Wf for this section?";WF
700 INPUT "dEu/dA for this section?";DEUDA
710 LET CMAF=CMAF+((WF^2)*DEUDA*DXF)
720 LET CM0F=CM0F+(WF^2)*(A0W+IFUS)*DXF
730 NEXT I
740 CMAF = CMAF/(36.5*SW*C)
750 CM0F = CM0F*DK/(36.5*SW*C)
760 LET CM0F=-.00158
770 LET CMAF=.04091
780 REM
790 REM
800 REM Find CM0w , CMaw , CM0t , CMat , CM0c , CMac
810 REM
820 CM0W = CMACW+CL0W*(XCG/C-XAC/C)
830 CMAW = CLAW*(XCG/C-XAC/C)
840 CM0T = ETA*VHT*(CLAT*(E0+IW-IT))
850 CM0C = VHC*(CLAC*(IW-IC))
860 CMAT =-ETA*VHT*(CLAT*(1-DEDA))
870 CMAC =-VHC*(CLAC)
880 REM
890 REM
900 REM Find CMw , CMf , CMt , CM0 , CMa , CMcg
910 CM0 = CM0W+CM0F+(ETA*VHT*CLAT)*(E0+IW-IT)+(ETA*VHC*CLAC)*(IW-IC)
930 CMA = CLAW*(XCG/C-XAC/C)+CMAF-(ETA*VHT*CLAT)*(1-DEDA)-(ETA*VHC*CLAC)
940 REM
960 A=.1745
980 CMW = CM0W+CMAW*A
990 CMF = CM0F+CMAF*A
1000 CMT = CM0T+CMAT*A
1010 CMC = CM0C+CMAC*A
1020 CMCG = CM0+(CMA*A)
1040 PRINT
1070 PRINT"CM0=";CM0
1080 PRINT"CMa=";CMA
1085 PRINT "CMa in non-linear (canard) region:";(CMA - CMAC)
1086 PRINT "CMa in non linear (tail) region:";(CMA - CMAT)
1090 REM
1100 REM Find Xnp
1110 REM
1120 XNP=C*(XAC/C-CMAF/CLAW+ETA*VHT*(CLAT/CLAW)*(1-DEDA)+VHC*CLAC/CLAW)
1130 PRINT "Xnp=";XNP
1133 PRINT"Stability Margin = ";((CMCG-CM0)/1.184)
1140 REM
1150 REM
1160 PRINT "Sample of points on curves at 0 AOA and 10 AOA"
1170 PRINT "COMPONENT:       0 DEGREES                      10 DEGREES"
1180 PRINT
1190 PRINT "Wing                ";CM0W;TAB(40)CMW
1200 PRINT "Canard              ";CM0C;TAB(40)CMC
1210 PRINT "Tail                ";CM0T;TAB(40)CMT
```

```
1220 PRINT "Fuselage          ";CMOF;TAB(40)CMF
1230 PRINT "Airplane          ";CM0;TAB(40)CMCG
1240 PRINT
1250 RUN
```

```
This version, STAT7.BAS, accepts degrees as input.

I  :? 0
Ic :? 0
Iw :? 0

CmO=-5.704538E-02
CMa=-.5551119
CMa in non-linear (canard) region:-1.442598
CMa in non linear (tail) region: .6425151
Xnp= 4.584932
Stability Margin = -8.181336E-02
Sample of points on curves at 0 AOA and 10 AOA
COMPONENT:        0 DEGREES                    10 DEGREES

Wing             -7.515221E-02                -.1250384
Canard            0                            .1548663
Tail              1.968683E-02                -.1652991
Fuselage          0                            5.558796E-03
Airplane         -5.704538E-02                -.1539124

This version, STAT7.BAS, accepts degrees as input.

It :?




This version, STAT7.BAS, accepts degrees as input.

It :? 2
Ic :? 0
Iw :? 0

CmO=-.1114826
CMa=-.5551119
CMa in non-linear (canard) region:-1.442598
CMa in non linear (tail) region: .6425151
Xnp= 4.584932
Stability Margin = -8.181336E-02
Sample of points on curves at 0 AOA and 10 AOA
COMPONENT:        0 DEGREES                    10 DEGREES

Wing             -7.515221E-02                -.1250384
Canard            0                            .1548663
Tail             -3.475035E-02                -.2437363
Fuselage          0                            5.558796E-03
Airplane         -.1114826                    -.2083496

This version, STAT7.BAS, accepts degrees as input.

It :?
```

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```
This version, STAT7.BAS, accepts degrees as input.

  :? 4
Ic :? 0
Iw :? 0

CmO=-.1659197
Cma=-.5551119
Cma in non-linear (canard) region:-1.442598
Cma in non linear (tail) region: .6425151
Xnp= 4.584932
Stability Margin = -8.181336E-02
Sample of points on curves at 0 AOA and 10 AOA
COMPONENT:          0 DEGREES                    10 DEGREES

Wing                -7.515221E-02                -.1250384
Canard               0                            .1548663
Tail                -8.918752E-02                -.2981734
Fuselage             0                            5.558796E-03
Airplane            -.1659197                    -.2627868

This version, STAT7.BAS, accepts degrees as input.

It :?
```

```
This version, STAT7.BAS, accepts degrees as input.

It :? 6
Ic :? 0
Iw :? 0

CmO=-.2203569
Cma=-.5551119
Cma in non-linear (canard) region:-1.442598
Cma in non linear (tail) region: .6425151
Xnp= 4.584932
Stability Margin = -8.181336E-02
Sample of points on curves at 0 AOA and 10 AOA
COMPONENT:          0 DEGREES                    10 DEGREES

Wing                -7.515221E-02                -.1250384
Canard               0                            .1548663
Tail                -.1436247                    -.3526106
Fuselage             0                            5.558796E-03
Airplane            -.2203569                    -.3172239

 s version, STAT7.BAS, accepts degrees as input.

It :?
```

Figure A.7.1
Wing Contribution to the
Pitching Moment

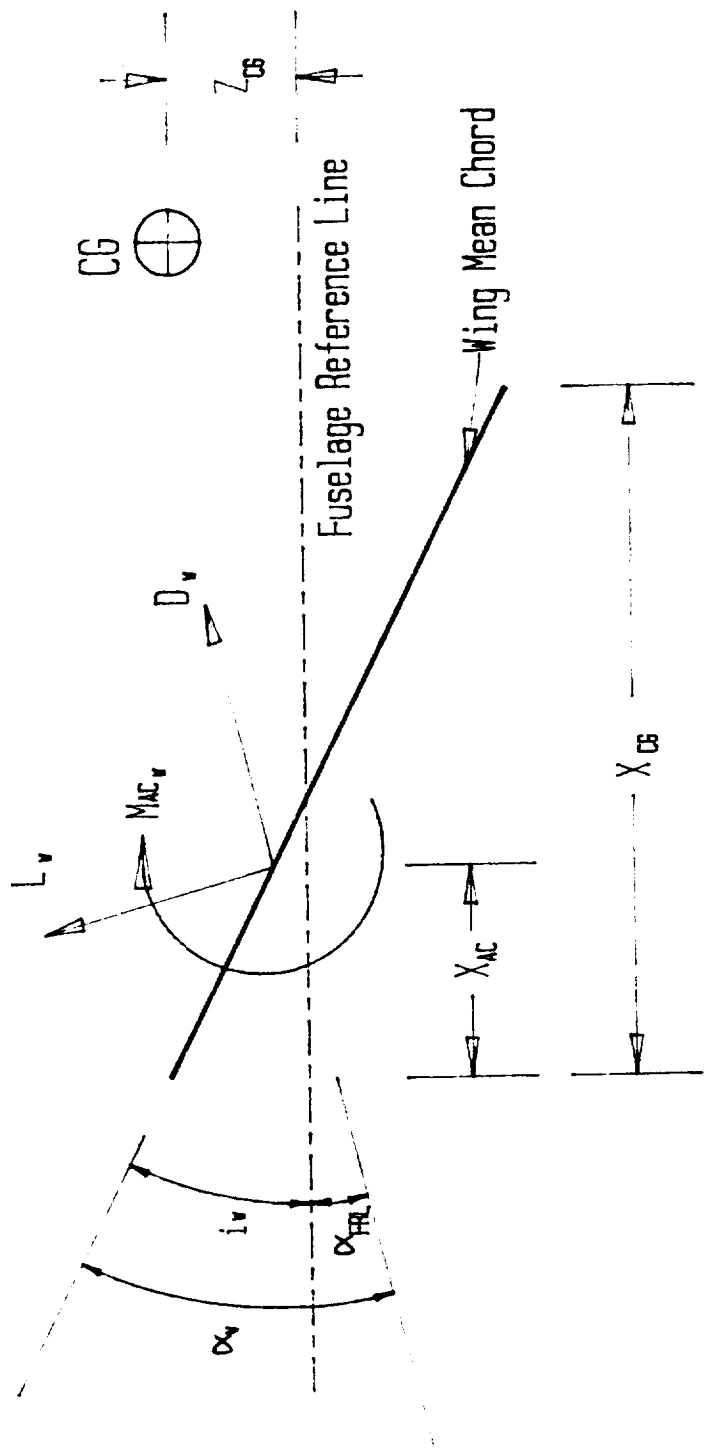
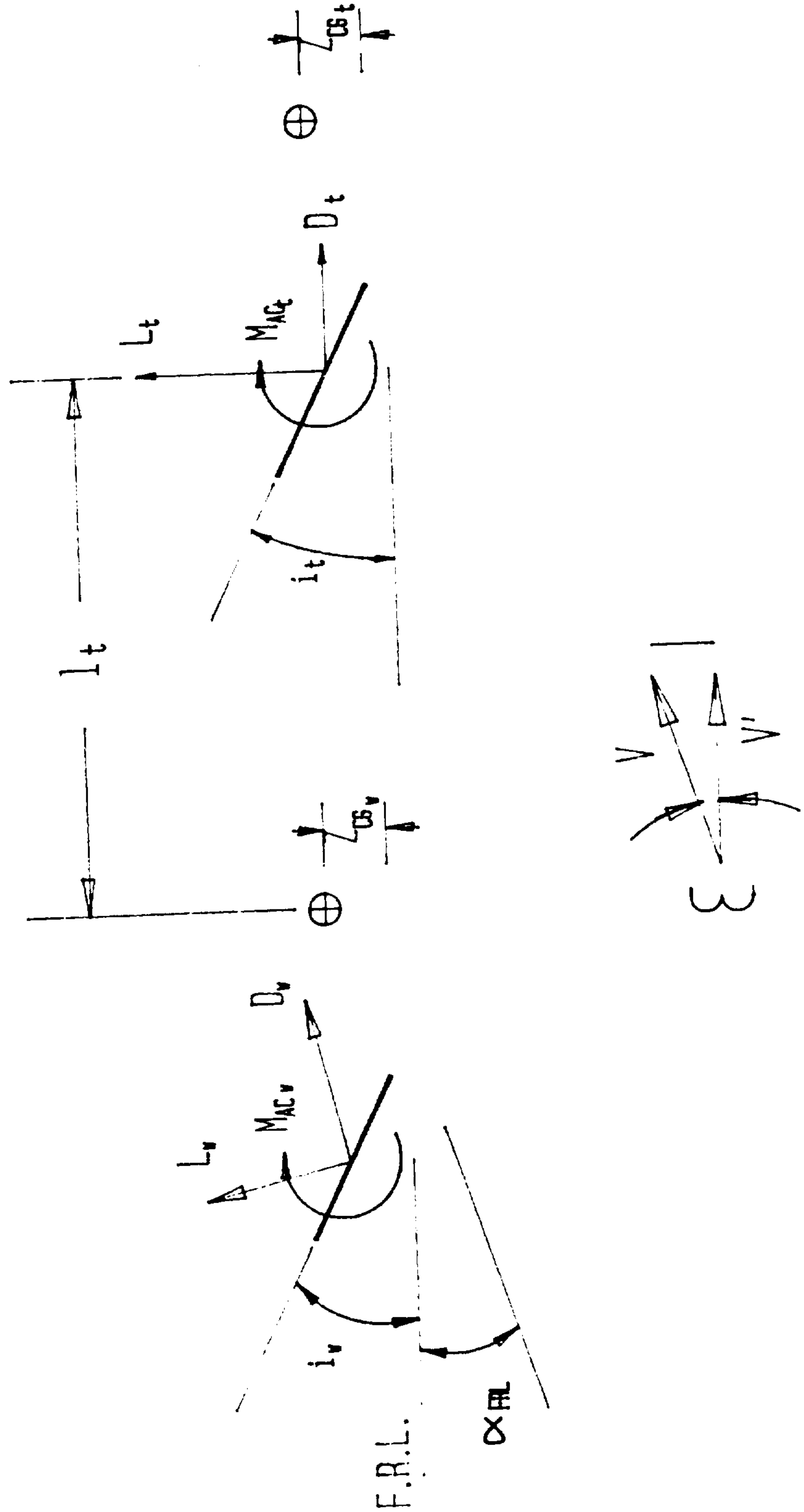
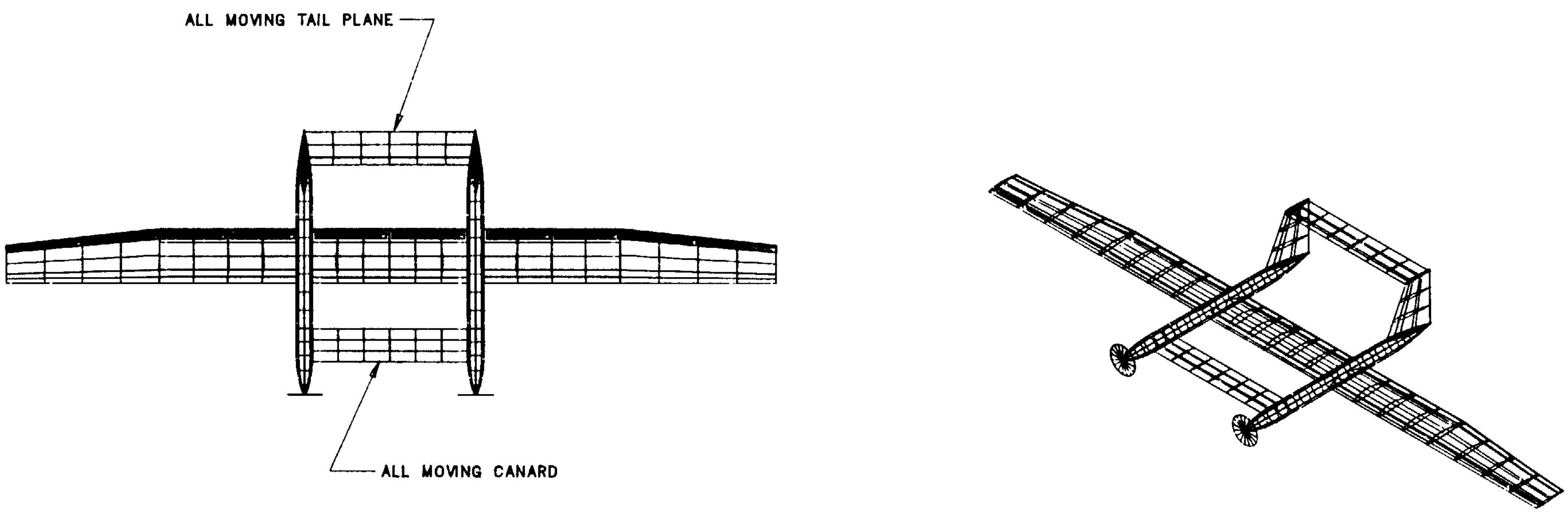


Figure A.7.2
Horizontal Tail Contribution to the
Pitching Moment





WPI CAD LABORATORY	
TITLE: CONTROL SURFACES	
DRAWN BY: TOM JUTRAS	
SCALE: 0.0025	DATE: 4/21/90
NO: 1	SHEET: 1

Appendix A.8

Dynamic Stability

The example on p. 131 of Reference 23 was used as a test case for the block diagram input into ASDEQ to solve equation 7.2.1. The theoretical approximations of the problem were compared to the results of the program (modified to show the long and short periods separately neglecting interactions). This comparison is summarized in the following table.

Theoretical approximations	Results of program*
Psp=2.26 sec	Psp=2.3 sec
Pp=25.54 sec	Pp=24.5 sec

*Extrapolated from Figure A.8.1 & A.8.2

The unmodified program, which takes the interconnectivity of the modes into account, was tested by solving the following equations simultaneously using data from Figure A.8.3. The graphical results are presented in Figures A.8.4 & A.8.5.

$$p\ p=-(1/T_d)\ln(u_2/u_1)$$
$$p(1-\ p^2)^{1/2}=2\ /T_d$$

The results compare to the example's exact results as follows.

Reference results　　Program results

p=.2144 rad/sec p=.21 rad/sec

The above comparisons validate the accuracy of both the main block diagram and the long and short period modifications.

Figure A.8.1

Short Period Mode

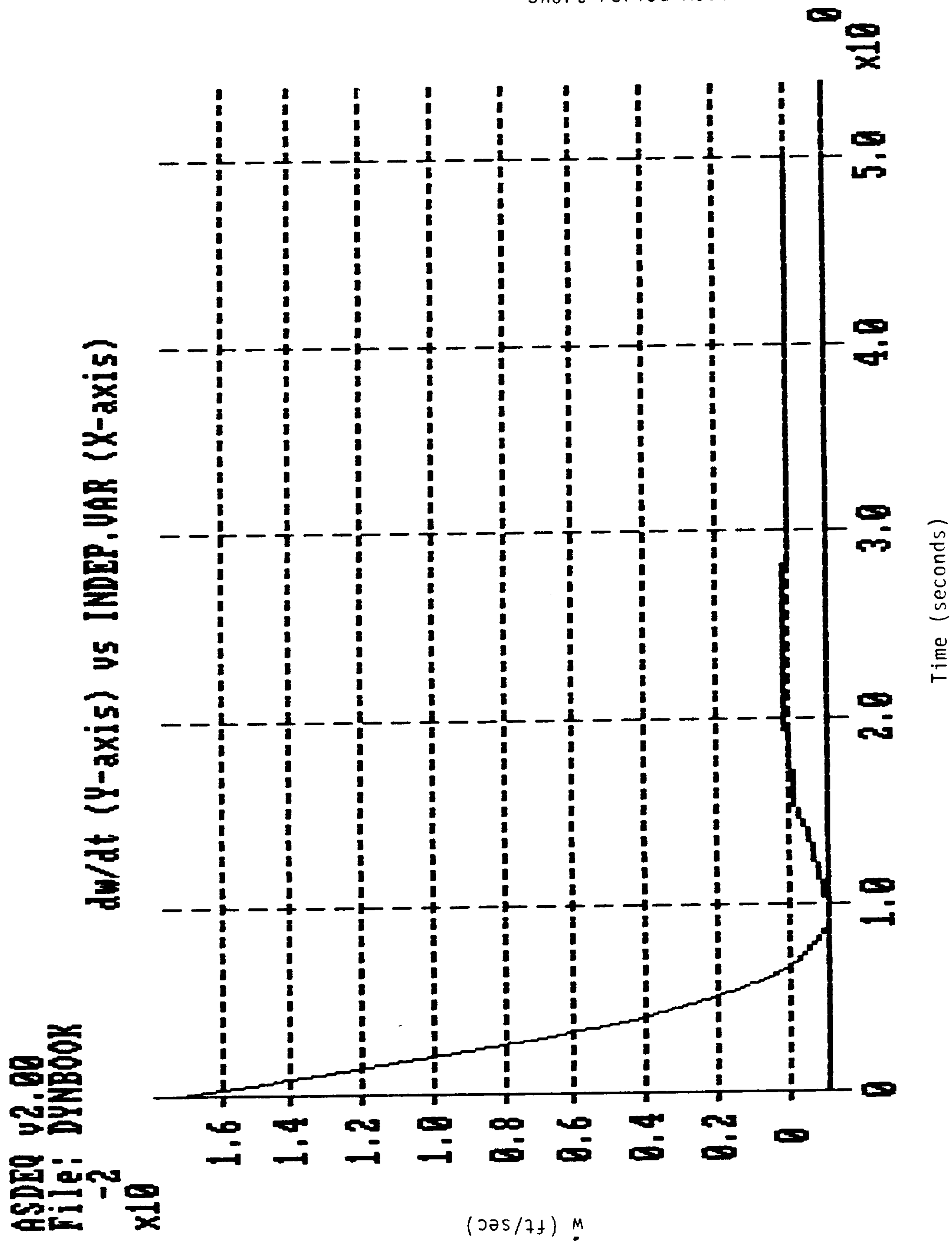


Figure A.8.2

Long Period Mode

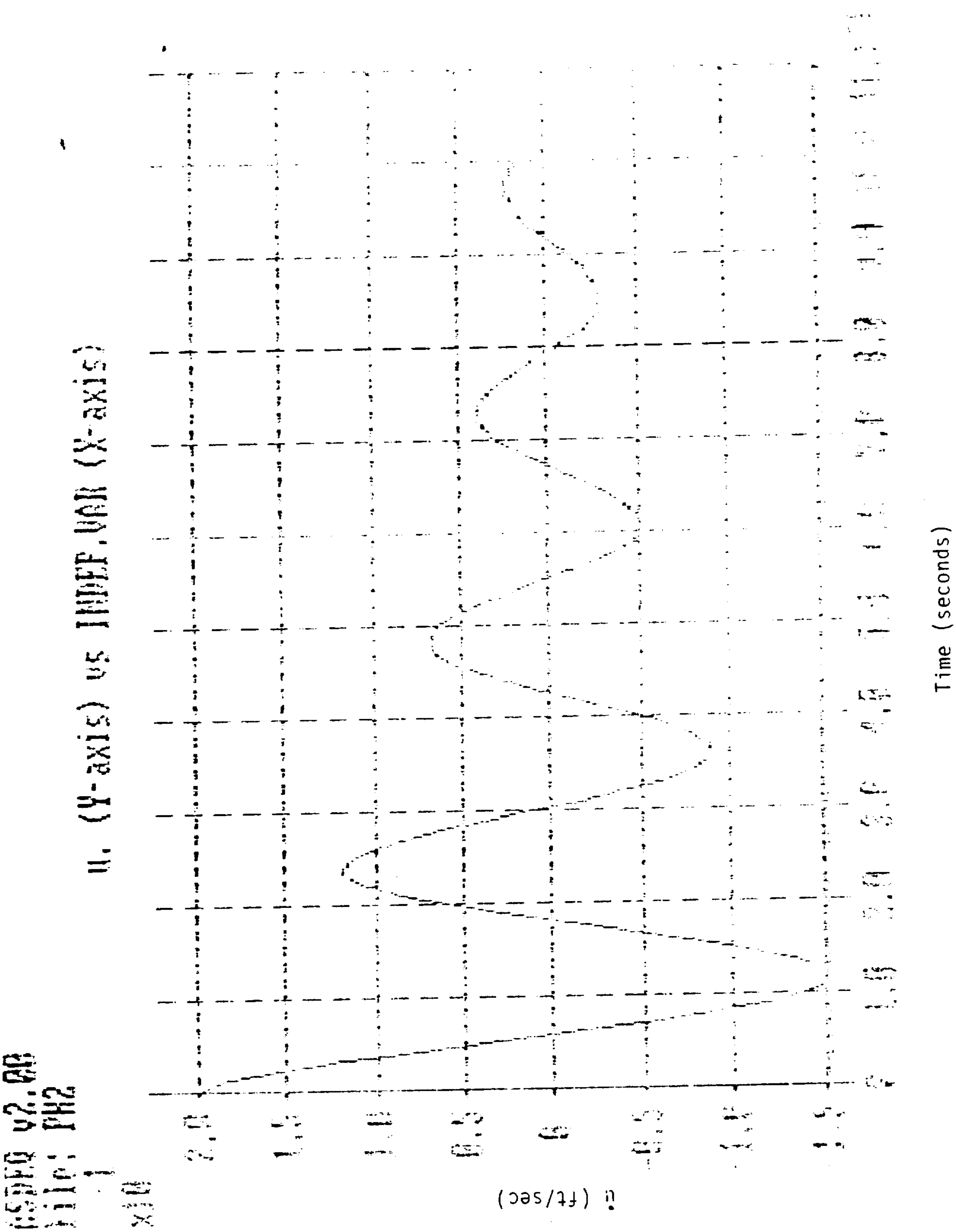


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Figure A.8.3

Combined Mode Data

Data for this run was read from A:DYNBOOK.ASD
under automatic step control with local error = .0005
on 04-17-1989 at 20:50:45

IND. VAR	BLOCK 1 du/dt	BLOCK 8 dw/dt	BLOCK 18 dq/dt
0.000	0	.017	0
0.500	3.888932E-04	.0021783	-6.7835E-05
1.000	1.131991E-03	-1.008656E-03	-1.044966E-05
1.500	1.958666E-03	-4.492623E-04	5.387086E-06
2.000	2.73988E-03	-1.519205E-04	4.652566E-06
2.500	3.467492E-03	-1.82749E-04	4.277175E-06
3.000	4.143304E-03	-2.456263E-04	5.170457E-06
3.500	4.762158E-03	-2.856293E-04	6.179634E-06
4.000	5.317657E-03	-3.159344E-04	7.034185E-06
4.500	5.804466E-03	-3.426886E-04	7.769967E-06
5.000	6.218296E-03	-3.656962E-04	8.4076E-06
5.500	6.555759E-03	-3.842862E-04	8.942817E-06
6.000	6.814365E-03	-3.982418E-04	9.369498E-06
6.500	6.992543E-03	-4.075071E-04	9.684288E-06
7.000	7.089645E-03	-4.120671E-04	9.885614E-06
7.500	7.105941E-03	-4.119495E-04	9.97317E-06
8.000	.0070426	-4.072329E-04	9.947877E-06
8.500	6.90166E-03	-3.980467E-04	9.811899E-06
9.000	6.685991E-03	-3.845682E-04	9.568606E-06
9.500	6.39925E-03	-3.670198E-04	9.222508E-06
10.000	6.045828E-03	-3.456654E-04	8.779188E-06
10.500	5.630785E-03	-3.208071E-04	8.24522E-06
11.000	5.159786E-03	-2.92781E-04	7.628083E-06
11.500	4.639028E-03	-2.61953E-04	6.936061E-06
12.000	4.075157E-03	-2.287144E-04	6.178132E-06
12.500	3.475193E-03	-1.934767E-04	5.363862E-06
13.000	2.846438E-03	-1.566668E-04	4.503289E-06
13.500	2.196396E-03	-1.18722E-04	3.606797E-06
14.000	1.532683E-03	-8.008524E-05	2.684998E-06
14.500	8.629402E-04	-4.119949E-05	1.748612E-06
15.000	1.947489E-04	-2.503266E-06	8.083396E-07
15.500	-4.644541E-04	3.557443E-05	-1.252517E-07
16.000	-1.107457E-03	7.261863E-05	-1.041847E-06
16.500	-1.727347E-03	1.082329E-04	-1.931492E-06
17.000	-2.317589E-03	1.420433E-04	-2.784702E-06
17.500	-2.872085E-03	1.737024E-04	-3.592555E-06
18.000	-3.385239E-03	2.028925E-04	-4.346787E-06
18.500	-3.852011E-03	2.293288E-04	-5.039868E-06
19.000	-4.267962E-03	2.527622E-04	-5.665082E-06
19.500	-4.629295E-03	2.729809E-04	-6.216578E-06
20.000	-4.932888E-03	2.898131E-04	-6.689427E-06
20.500	-5.176314E-03	3.031271E-04	-7.079664E-06
21.000	-5.357859E-03	3.128331E-04	-7.38431E-06
21.500	-5.476531E-03	3.188829E-04	-7.601391E-06
22.000	-5.532053E-03	3.212701E-04	-7.729946E-06
22.500	-5.524861E-03	3.200293E-04	-7.770021E-06
23.000	-5.456083E-03	3.152348E-04	-7.722642E-06
23.500	-5.327515E-03	3.069996E-04	-7.589803E-06
24.000	-5.141594E-03	2.954731E-04	-7.374414E-06
24.500	-4.901353E-03	2.808391E-04	-7.080258E-06
25.000	-4.610383E-03	2.63313E-04	-6.711941E-06
25.500	-4.272782E-03	2.431387E-04	-6.27481E-06
26.000	-3.893099E-03	2.20586E-04	-5.774901E-06
26.500	-3.476278E-03	1.959464E-04	-5.218845E-06

[illegible]	[illegible]	[illegible]	[illegible]
28.000	-2.057018E-03	1.12676E-04	-3.287328E-06
28.500	-1.546747E-03	8.291658E-05	-2.581986E-06
29.000	-1.027726E-03	5.272829E-05	-1.859591E-06
29.500	-5.058981E-04	2.245542E-05	-1.128495E-06
30.000	1.286214E-05	-7.562486E-06	-3.970101E-07
30.500	5.228119E-04	-3.699442E-05	3.266886E-07
31.000	1.018403E-03	-6.552146E-05	1.034648E-06
31.500	1.494339E-03	-9.284009E-05	1.719223E-06
32.000	1.945634E-03	-1.186654E-04	2.37316E-06
32.500	2.367659E-03	-1.427339E-04	2.989666E-06
33.000	2.75619E-03	-1.648061E-04	3.56248E-06
33.500	3.107445E-03	-1.846688E-04	4.085936E-06
34.000	3.418126E-03	-2.021372E-04	4.555011E-06
34.500	3.685439E-03	-2.170561E-04	4.965376E-06
35.000	3.907125E-03	-2.293015E-04	5.313427E-06
35.500	4.081468E-03	-2.387812E-04	5.596318E-06
36.000	4.207314E-03	-2.454357E-04	5.811979E-06
36.500	4.284067E-03	-2.492378E-04	5.959124E-06
37.000	4.311689E-03	-2.501926E-04	6.037254E-06
37.500	4.290693E-03	-2.483372E-04	6.046649E-06
38.000	4.222125E-03	-2.437391E-04	5.988353E-06
38.500	4.107547E-03	-2.364955E-04	5.864149E-06
39.000	3.949007E-03	-2.267316E-04	5.676523E-06
39.500	3.749009E-03	-2.145984E-04	5.428632E-06
40.000	3.510481E-03	-2.002711E-04	5.12425E-06
40.500	3.236729E-03	-1.839465E-04	4.767723E-06
41.000	2.931401E-03	-1.6584E-04	4.363901E-06
41.500	2.598434E-03	-1.461839E-04	3.918087E-06
42.000	2.242008E-03	-1.252234E-04	3.43596E-06
42.500	1.866495E-03	-1.032145E-04	2.923512E-06
43.000	1.476411E-03	-8.042053E-05	2.386974E-06
43.500	1.076353E-03	-5.710918E-05	1.832739E-06
44.000	6.709574E-04	-3.354944E-05	1.267295E-06
44.500	2.648409E-04	-1.000859E-05	6.971433E-07
45.000	-1.374485E-04	1.325074E-05	1.287307E-07
45.500	-5.314818E-04	3.597348E-05	-4.316231E-07
46.000	-9.12996E-04	5.791481E-05	-9.777946E-07
46.500	-1.277939E-03	7.884274E-05	-1.50392E-06
47.000	-1.622512E-03	9.854043E-05	-2.004459E-06
47.500	-1.943207E-03	1.168085E-04	-2.474248E-06
48.000	-2.23684E-03	1.334667E-04	-2.908553E-06
48.500	-2.500585E-03	1.483559E-04	-3.303118E-06
49.000	-2.731995E-03	1.613394E-04	-3.654198E-06
49.500	-2.929025E-03	1.72304E-04	-3.9586E-06
50.000	-3.09005E-03	1.81161E-04	-4.213703E-06
50.500	-3.213872E-03	1.878467E-04	-4.417482E-06
51.000	-3.299731E-03	1.923227E-04	-4.568519E-06
51.500	-.0033473	1.945758E-04	-4.666007E-06
52.000	-3.35669E-03	1.946181E-04	-4.709752E-06
52.500	-3.328432E-03	1.924861E-04	-4.700164E-06
53.000	-3.263472E-03	1.882398E-04	-4.638241E-06
53.500	-3.163148E-03	1.81962E-04	-4.525548E-06
54.000	-3.029171E-03	1.737568E-04	-4.364193E-06
54.500	-.0028636	1.637481E-04	-4.156791E-06
55.000	-2.668812E-03	1.520777E-04	-3.906428E-06
55.500	-2.447471E-03	1.389037E-04	-3.616617E-06
56.000	-2.202493E-03	1.243984E-04	-3.291254E-06
56.500	-1.937008E-03	1.087461E-04	-2.934567E-06
57.000	-1.654324E-03	9.214067E-05	-2.551064E-06
57.500	-1.357886E-03	7.478359E-05	-2.145476E-06
58.000	-1.051235E-03	5.688127E-05	-1.722701E-06
58.500	-7.379656E-04	3.864272E-05	-1.287749E-06
59.000	-4.216879E-04	2.02773E-05	-8.45681E-07
59.500	-1.059846E-04	1.992225E-06	-4.015534E-07

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61.500	1.08254E-03	-6.639754E-05	1.297714E-06
62.000	1.345246E-03	-8.139913E-05	1.680314E-06
62.500	1.588538E-03	-9.524095E-05	2.037754E-06
63.000	1.810024E-03	-1.077878E-04	2.366464E-06
63.500	2.007593E-03	-1.189211E-04	2.663251E-06
64.000	2.179439E-03	-1.285403E-04	2.925334E-06
64.500	2.324072E-03	-1.36563E-04	3.150366E-06
65.000	2.440332E-03	-1.429265E-04	3.336453E-06
65.500	2.527393E-03	-1.475874E-04	3.482166E-06
66.000	2.584772E-03	-1.50522E-04	3.586552E-06
66.500	2.612324E-03	-1.517264E-04	3.649138E-06
67.000	2.610238E-03	-1.512161E-04	3.669921E-06
67.500	2.579034E-03	-1.490254E-04	3.649368E-06
68.000	2.519547E-03	-1.452069E-04	3.5884E-06
68.500	2.432913E-03	-1.398303E-04	3.488371E-06
69.000	2.320553E-03	-1.329819E-04	3.351053E-06
69.500	2.184153E-03	-1.247626E-04	3.178601E-06
70.000	2.025637E-03	-1.152872E-04	2.973529E-06
70.500	1.847145E-03	-1.046824E-04	2.738672E-06
71.000	1.651005E-03	-9.308567E-05	2.477152E-06
71.500	1.439702E-03	-8.0643E-05	2.192334E-06
72.000	1.215849E-03	-6.750751E-05	1.887788E-06
72.500	9.82157E-04	-5.383751E-05	1.567246E-06
73.000	7.413993E-04	-3.979458E-05	1.234554E-06
73.500	4.963834E-04	-2.554181E-05	8.93633E-07
74.000	2.49917E-04	-1.124188E-05	5.484263E-07
74.500	4.776952E-06	2.94469E-06	2.028618E-07
75.000	-2.363209E-04	1.686136E-05	-1.391951E-07
75.500	-4.707513E-04	3.035721E-05	-4.739827E-07
76.000	-6.960074E-04	4.328852E-05	-7.978833E-07
76.500	-9.097265E-04	5.552031E-05	-1.107461E-06
77.000	-1.109715E-03	6.692766E-05	-1.399497E-06
77.500	-1.293967E-03	7.73969E-05	-1.671024E-06
78.000	-1.460689E-03	8.68268E-05	-1.919351E-06
78.500	-1.608312E-03	9.512943E-05	-2.142092E-06
79.000	-1.735505E-03	1.022308E-04	-2.337188E-06
79.500	-1.841189E-03	1.080718E-04	-2.50292E-06
80.000	-1.924544E-03	1.126081E-04	-2.637928E-06
80.500	-1.985012E-03	1.15811E-04	-2.741218E-06
81.000	-2.022299E-03	1.176669E-04	-2.812167E-06
81.500	-2.036376E-03	1.181775E-04	-2.850521E-06
82.000	-2.027474E-03	1.173598E-04	-2.856398E-06
82.500	-1.996076E-03	1.152449E-04	-2.830273E-06
83.000	-1.942909E-03	1.118782E-04	-2.772975E-06
83.500	-1.86893E-03	1.073182E-04	-2.685662E-06
84.000	-1.775314E-03	1.016357E-04	-2.569813E-06
84.500	-1.663433E-03	9.491306E-05	-2.427197E-06
85.000	-1.534843E-03	8.724264E-05	-2.259856E-06
85.500	-1.391258E-03	7.872596E-05	-2.070069E-06
86.000	-1.234534E-03	6.947233E-05	-1.860333E-06
86.500	-1.06664E-03	5.959733E-05	-1.633324E-06
87.000	-8.896376E-04	4.922169E-05	-1.391865E-06
87.500	-7.056562E-04	3.846965E-05	-1.138894E-06
88.000	-5.168671E-04	2.746759E-05	-8.774296E-07
88.500	-3.254594E-04	1.634254E-05	-6.105334E-07
89.000	-1.336145E-04	5.220841E-06	-3.412773E-07
89.500	5.651762E-05	-5.773366E-06	-7.270802E-08
90.000	2.42842E-04	-1.651941E-05	1.921864E-07
90.500	4.233411E-04	-2.690142E-05	4.505087E-07
91.000	5.960965E-04	-3.680951E-05	6.994837E-07
91.500	7.593079E-04	-4.614096E-05	9.364866E-07
92.000	9.113121E-04	-5.480116E-05	1.15907E-06
92.500	1.050598E-03	-6.270462E-05	1.36499E-06

93.000	1.175824E-03	-6.977568E-05	1.552224E-06
93.500	1.285824E-03	-7.594932E-05	1.718994E-06
94.000	1.379626E-03	-8.117152E-05	1.863777E-06
94.500	1.456451E-03	-8.539992E-05	1.985326E-06
95.000	1.515727E-03	-8.860382E-05	2.082671E-06
95.500	1.557083E-03	-9.07646E-05	2.155128E-06
96.000	1.580357E-03	-9.187555E-05	2.202304E-06
96.500	1.585591E-03	-9.194179E-05	2.224094E-06
97.000	1.573029E-03	-9.098003E-05	2.220681E-06
97.500	1.543108E-03	-8.901821E-05	2.192521E-06
98.000	1.496452E-03	-8.609494E-05	2.140345E-06
98.500	1.433862E-03	-8.225893E-05	2.065134E-06
99.000	1.356303E-03	-7.756826E-05	1.968116E-06
99.500	1.264892E-03	-7.20896E-05	1.850736E-06
100.000	1.16088E-03	-6.589732E-05	1.714647E-06

Exact solution of example problem - proof of program function

Figure A.8.4
Combined Mode

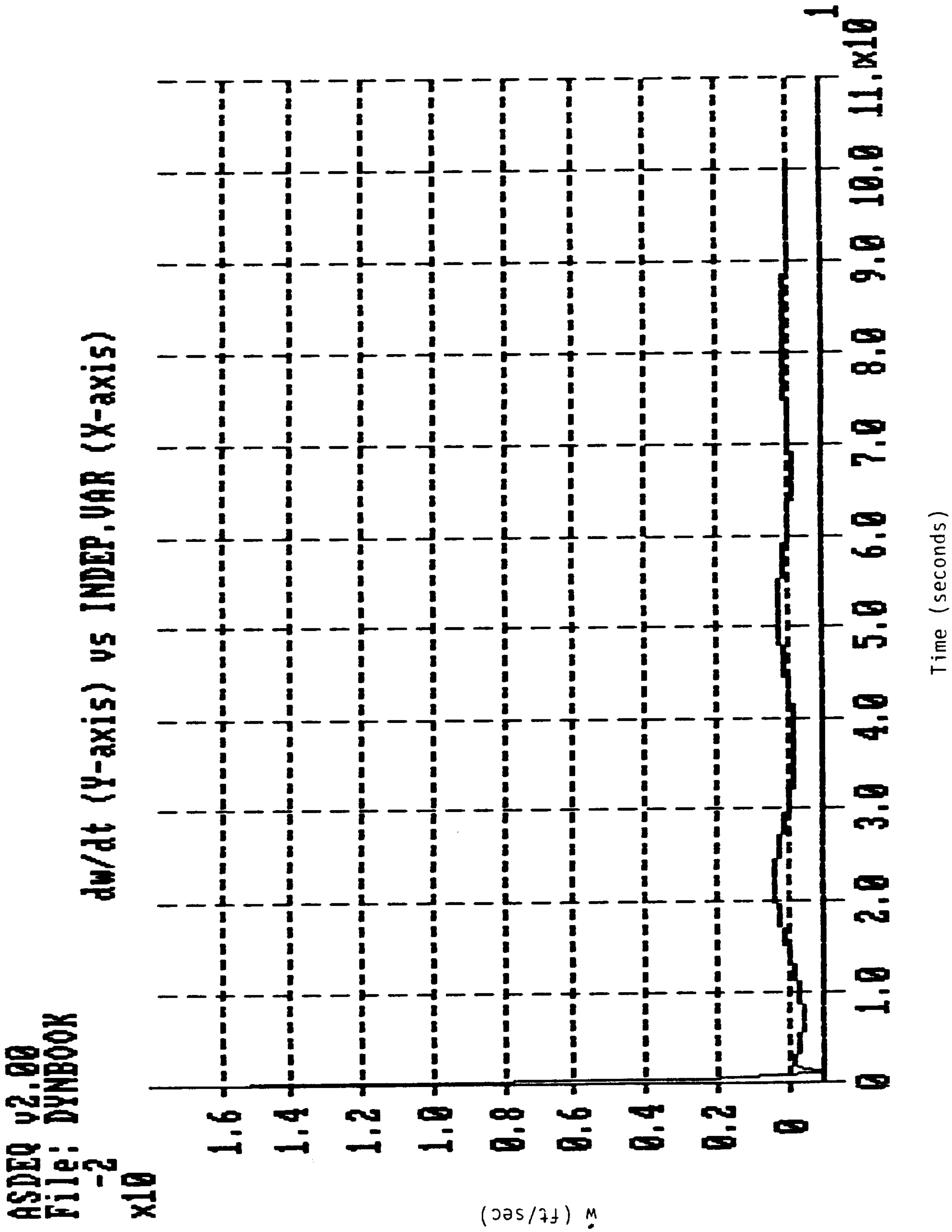
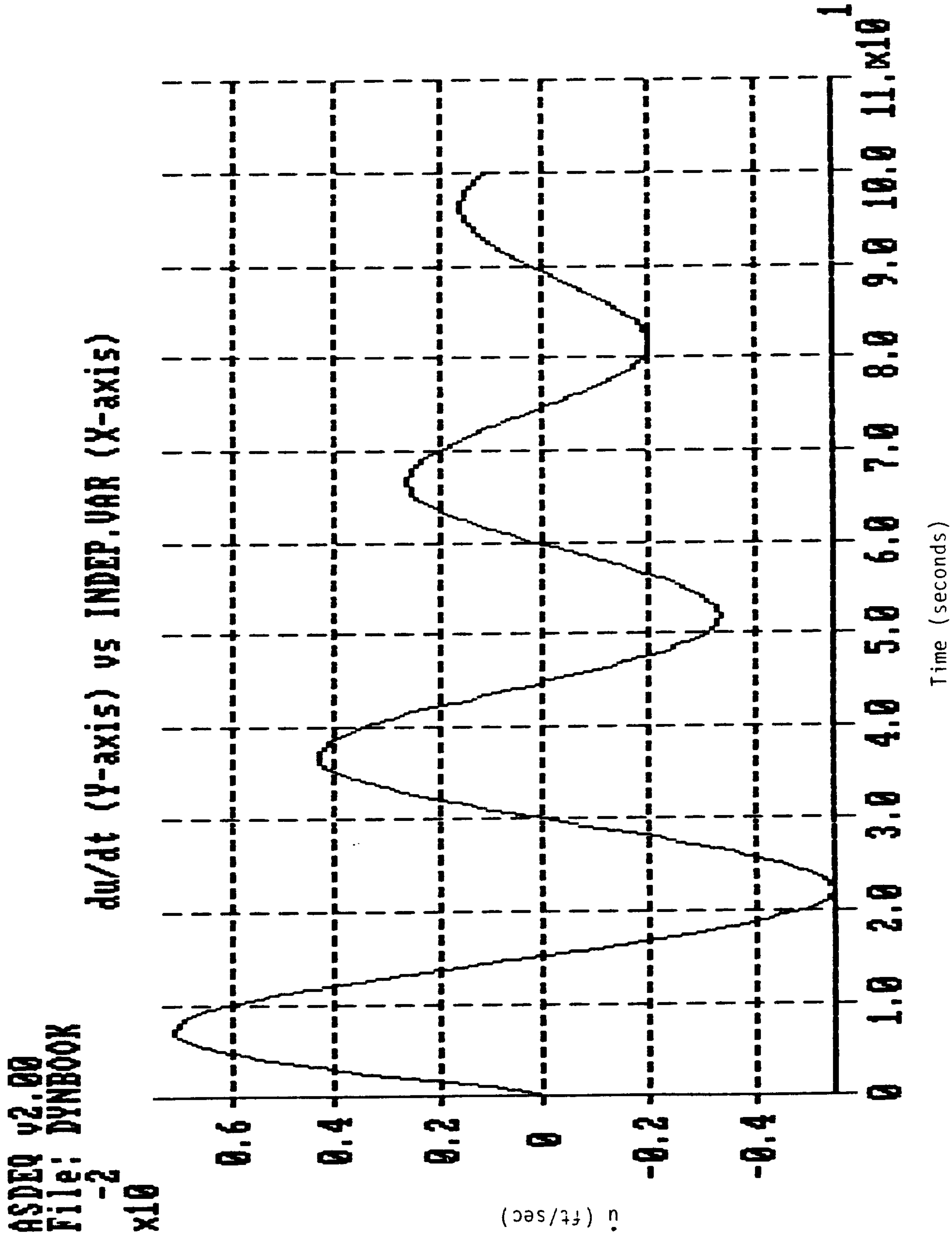


Figure A.8.5
Combined Mode



Appendix A.9

Performace Calculations

Table of Contents

9.2 Power Analysis

9.2.1 Level Flight

Level flight power analysis program

Lotus spreadsheet - Power required & available

9.2.2 Turning Flight

Turning flight power analysis BASIC program

Lotus spreadsheet (sample - bank angle of 15°)

9.3 Climb Performance

9.3.1 Climb Rate

Lotus spreadsheet (sample - bank angle Of 10°)

9.4 Flight Path

Lotus spreadsheets (sample - 50,000 feet)

Final path description

Number of turns to altitude

9.5 V-n Diagram

Wind gust program

Lotus spreadsheet

9.2 Power Analysis

9.2.1 Level Flight

```
100 REM          Power Analysis
110 REM
120 REM          Level Flight
130 REM
140 REM ******* Variables and Constants *******
150 REM
160 REM          M  - Mach Number
170 REM          S  - Wing Area  (ft^2)
180 REM          RO - Density  (lb/ft^3)
190 REM          V  - Velocity  (ft/s)
200 REM          AR - Aspect Ratio
210 REM          FE - Bank Angle  (degrees)
220 REM          NF - Bank angle (radians)
230 REM          L  - Lift  (lb)
240 REM          W  - Weight  (lb)
250 REM          CD - Coefficient of Drag
260 REM          CO - Coefficient of Drag at zero lift
270 REM          K  - Constant
280 REM          CL - Coefficient of Lift
290 REM          PI - 3.14
300 REM          P  - Power  (hp)
310 REM          D  - Drag
320 REM          E  - Airplane efficiency
330 REM          PA - Power available (can be generated by the motor)
340 REM          RG - Power generated/power received by motor
350 REM          EM - Motor efficiency
360 REM          EP - Propeller efficiency
370 REM          PD - Power Density (hp/ft^2)
380 REM          FA - Altitude (ft)
390 REM          PR - Power required (must be supplied to the motor)
400 REM          T  - Thrust
410 REM          FR - Flight radius
420 REM          AI - Angle of incidence
430 REM          AD - Angle of incidence (degrees)
440 REM
450 REM ******* Define Constants *******
460 REM
470              E = .85              :REM  Airplane efficiency
480              PI = 3.1416          :REM  Value of one radian
490              I = .01              :REM  Increase the Mach #
500              ST = 5               :REM  Increase the bank angle
510              K = .007             :REM  Viscous Drag Coefficient estimated
                                            from a NACA 2214 airfoil
520              G = 32.174           :REM  Gravitational constant  (ft/s^2)
530              RG = .97             :REM  Power recieved/power generated
540              EM = .98             :REM  Motor efficiency
550              AP = 175             :REM  Propeller area  (ft^2)
560              CP = 550             :REM  Conversion from hp to ft*lbf/s
570              PD = (700*1.3405*10^-3*(.3048)^2)
580 REM
590 REM ******* Data Entry *******
600 REM
610              INPUT "Gross Weight";W
620              INPUT "Wing Area"; S
630              INPUT "Aspect Ratio";AR
640              INPUT "Altitude of flight";FA
650              INPUT "Speed of sound at altitude";SS
660              INPUT "Density at altitude";RO
670              INPUT "Coefficient of Drag @ zero lift";CO
680              INPUT "Total propeller area";AP
690 REM
700 REM ******* Print Input Constants *******
```

```
720           PRINT "Gross Weight                                      ";W
730           PRINT "Wing Area                                         ";S
740           PRINT "Aspect Ratio                                      ";AR
750           PRINT "Altitude of flight                                ";FA
760           PRINT "Speed of sound @ altitude                         ";SS
770           PRINT "Density at altitude                               ";RO
780           PRINT "Coefficient of Drag @ zero lift                   ";CO
              PRINT "Total propeller area                              ";AP
800 REM
810 REM ******* Calculations *******
820 REM
830           OPEN "A:100CR2.out" FOR OUTPUT AS 1 LEN=2000
840 REM
850           L = W
860           FOR M = .01 TO 1.1 STEP I
870 REM
880                  V = M*SS                        :REM VELOCITY
890 REM
900                  CL = (L/(.5*RO*S*(V^2))):REM COEFF OF
                                                   LIFT
910                  CD = (CO + K*(CL^2) + (CL^2)/(E*PI*AR) )
                                                  :REM COEFF OF
                                                   DRAG
920                  D = CD*.5*RO*(V^2)*S         :REM DRAG
930 REM
940                  P = D*V/CP                   :REM POWER REQUIRED (hp)
950                  PR = PD*S                    :REM POWER RECEIVED
960                  PA = RG*EM*EP*PR             :REM POWER AVAILABLE
970 REM
980                  T = ((PR/V)*CP)              :REM THRUST
990 REM
1000                  EP = (2/(1+(1+(T/(.5*RO*(V)^2*AP)))^.5))
                                                   :REM PROP EFFICIENCY
1010 REM
1020                  PA = RG*EM*EP*PR                   :REM POWER AVAILABLE
1030                  DH = ((PA - P)/W)*CP               :REM RATE OF CLIMB
1040 REM
1050                  PRINT "MACH #";M;"   CLIMB RATE   ";DH
1060                  PRINT#1, M;DH
1070 REM
1080           NEXT M
1090 REM
1100           CLOSE 1
1110 REM
1120           END
```

Power Available (Bank Angle = 0)

Mach #	Altitude (feet) Sea Lev	25,000	50,000	75,000	100,000
0.01	22.95517	13.9269	7.955211	4.240042	2.48042
0.02	51.22117	37.7642	21.1425	11.8262	6.722031
0.03	103.3297	60.59135	37.89791	21.25825	12.24174
0.04	142.512	94.26285	56.9742	32.2789	18.92425
0.05	175.7255	123.2007	75.1125	44.10759	25.97031
0.06	204.437	149.0651	94.42281	56.52902	33.80139
0.07	225.452	172.553	112.3955	69.37535	41.77022
0.08	243.3292	193.2726	121.6292	82.25757	50.19746
0.089999	255.1462	211.3222	142.5752	95.21751	58.58537
0.099999	265.512	225.7737	164.9224	108.1117	67.86705
0.11	277.194	221.1455	172.5722	121.2137	76.25472
0.12	272.5574	248.0941	183.1460	133.7404	85.45301
0.13	282.9210	255.7712	205.1791	144.3953	94.22450
0.14	290.2105	253.9779	212.7814	155.5278	103.207
0.15	222.7814	259.552	227.7112	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	295.4216	277.3584	241.2227	180.4527	122.8447
0.18	293.7072	281.1024	248.2242	194.2857	127.0119
0.19	294.7939	282.7796	254.0907	202.4505	144.9427
0.2	295.5771	285.0162	259.1754	210.0922	152.6305
0.21	288.2907	287.3012	262.0570	217.0918	160.0572
0.22	296.9916	289.4972	267.462	227.2945	167.2202
0.23	297.4552	290.2542	270.5219	225.5881	174.1039
0.240000	297.9228	292.0129	272.9481	235.1025	180.6991
0.25	298.2868	293.0097	276.5057	240.1985	187.0092
0.26	298.5592	293.2689	278.5257	244.8173	192.0344
0.27	298.2825	294.8125	281.0373	248.0755	198.7772
0.28	291.0282	295.2612	282.2775	252.8247	204.2415
[illegible]	197.2912	297.4275	[illegible]	255.7004	[illegible]
0.3	292.4712	286.2245	282.1742	259.5171	214.2116
0.31	299.2792	295.7522	287.1874	262.8258	219.0262
[illegible]	[illegible]	297.1402	[illegible]	265.2848	222.441
[illegible]	[illegible]	287.4917	290.4755	266.1118	227.5212
0.34	299.2276	297.7257	290.312	270.409	221.5676
0.349999	300.0522	298.0539	291.2259	272.9252	235.2972
0.359999	300.1222	299.2128	291.9814	274.4076	232.2072
0.369999	300.2422	287.3212	293.124	275.2514	242.1012
0.379999	300.7102	296.722	293.2221	277.331	242.2442
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.399999	299.4271	297.0011	294.2222	280.7821	247.3271
[illegible]	[illegible]	[illegible]	[illegible]	282.152	[illegible]
0.419999	300.5217	297.7442	275.2402	282.2402	252.062
0.429999	300.5741	299.4247	291.7274	284.3392	252.2223
0.439999	300.2012	298.2747	292.2219	285.347	250.5102
0.449999	300.2217	299.4792	295.307	286.2752	262.559
0.459999	300.2071	299.7274	296.9052	287.1422	264.4272
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	267.1157
0.489999	300.7702	300.1272	297.2241	290.1198	271.1435

Level Flight Analysis

	300.7911	300.1362	297.7944	290.5175	272.3741
0.509999	300.8104	300.1847	297.9699	291.1745	273.923
0.519999	300.8292	300.237	298.1421	291.5939	275.195
0.529999	300.8448	300.2856	298.3021	292.1789	276.3951
0.539999	300.8601	300.3305	298.451	292.6381	277.5277
0.549999	300.8744	300.3728	298.5897	293.026	278.5371
0.559999	300.8877	300.4117	298.7191	293.452	279.5072
0.569999	300.9001	300.4402	298.8399	293.8242	280.5517
0.579999	300.9117	300.4922	298.9528	294.1725	281.4542
0.589999	300.9222	300.514	299.0526	294.5011	282.3177
0.599999	300.9322	300.5433	299.1575	294.8037	283.1257
0.609999	300.9401	300.5717	299.2504	295.0879	283.8896
0.619999	300.951	300.5979	299.3375	295.3701	284.514
0.629999	300.9593	300.6224	299.4194	295.6264	285.2002
0.639999	300.9672	300.6455	299.4954	295.858	285.9507
0.649999	300.9745	300.6672	299.5662	296.095	286.5677
0.659999	300.9813	300.6877	299.6271	296.3111	287.1532
0.669999	300.988	300.7069	299.7018	296.5144	287.7001
0.679999	300.9946	300.7251	299.7582	296.7055	288.2271
0.689999	301	300.7422	299.8125	296.8882	288.7288
0.699999	[illegible]	300.7584	299.8731	297.0502	289.2139
[illegible]	[illegible]	[illegible]	299.9221	297.2234	289.6597
0.719999	301.0155	300.7882	299.9727	297.3779	290.1015
0.729999	301.0203	300.802	300.0157	297.5245	290.5125
0.739999	301.0247	300.812	300.0522	297.6627	290.9042
0.749999	301.0289	300.8273	300.1046	297.7989	291.2774
0.759999	301.0328	300.839	300.1425	297.9214	291.6322
0.769999	301.0362	300.8501	300.1812	298.0408	291.9729
0.779999	301.0402	300.8607	300.2155	298.1545	292.2955
0.789999	301.0425	300.8707	300.2503	298.2627	292.6057
0.799999	301.0460	300.8802	300.2824	298.3557	292.901
0.809999	301.0489	300.8893	300.3129	298.4629	293.1822
0.819999	301.0522	300.898	300.342	298.5575	293.452
0.829999	[illegible]	300.9062	300.3698	298.647	293.7109
0.839999	[illegible]	[illegible]	[illegible]	298.7332	[illegible]
0.849999	301.0598	300.9219	300.4215	298.8132	294.1933
0.859999	301.0622	300.9292	300.4456	298.8915	294.4202
0.869999	301.0646	300.9362	300.4685	298.9661	294.6285
0.879999	301.0672	300.9431	300.4905	299.0372	294.8444
0.889999	301.0699	300.9484	300.5117	299.1054	295.0425
0.899999	301.0719	300.9544	300.5319	299.1705	295.2244
0.909999	301.0739	300.9601	300.5511	299.2221	295.4175
0.919999	301.0757	300.9656	300.5695	299.292	295.5922
0.929999	301.0776	300.9708	300.5872	299.3604	295.7621
0.939999	[illegible]	300.9758	300.6043	299.4054	295.9142
0.949999	[illegible]	300.9807	300.6205	299.4562	296.06
0.959999	[illegible]	300.9852	300.6361	[illegible]	296.2029
0.969999	[illegible]	300.9899	300.6511	299.5572	296.3726
0.979999	[illegible]	300.994	300.6654	299.6042	296.512
0.989999	301.0867	300.9981	300.6793	299.6492	296.6452
0.999999	[illegible]	301.0021	300.6926	299.6924	296.7724
[illegible]	301.0892	[illegible]	300.7053	299.734	296.8962
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]					

LEVEL FLIGHT ANALYSIS

1.059999	301.0951	301.0227	300.7533	299.9194	297.4492
1.079999	301.0951	301.0227	300.7724	299.9526	297.5485
1.089999	301.097	301.0295	300.7822	299.9845	297.644
1.099999	301.098	301.0214	300.7915	300.0152	297.7362

Level Flight Summary

	Power Required (Bank Angle = 0)				
	Altitude (feet)				
Mach #	Sea Lev	25,000	50,000	75,000	100,000
	24.60831	122.3445	405.5074	1351.65	4488.105
0.01	21.74814	57.16057	202.0373	672.8887	2241.[illegible]
0.02	22.22347	45.13232	137.148	450.7123	1404.084
0.03	25.31[illegible]1	27.42428	102.7012	336.2937	1120.646
0.04	24.62272	24.7384	84.37251	271.0549	898.5507
0.05	50.83423	35.26735	72.4824	226.3267	747.4241
0.06	72.77751	41.35[illegible]	66.14241	195.0954	640.5471
0.07	103.327	45.5745	61.06574	171.9428	561.2253
0.08	145.239	52.64347	59.54494	154.4221	495.2353
0.089999	197.3702	71.23502	60.12567	140.9928	450.0324
0.099999	261.[illegible]	91.[illegible]	62.52259	130.6226	408.[illegible]
0.11	337.[illegible]	122.5442	67.24572	122.7043	376.7551
0.12	[illegible]5.5774	151.2062	73.7273	116.7716	348.5112
0.13	531.351	152.7682	81.58133	112.5203	325.3262
0.14	647.0825	221.4576	92.148	109.7354	305.1722
0.15	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.16	[illegible]	[illegible]	[illegible]	[illegible]	272.3105
0.17	1128.258	322.0817	124.736	109.0511	260.0071
0.18	1322.753	452.217	133.2129	111.0904	248.6335
0.19	1548.13	527.608	174.1450	114.1759	229.1353
0.2	1784.705	602.5425	197.3802	118.2051	220.9024
0.21	2051.424	695.3028	223.1455	123.4577	212.3134
0.22	2342.551	787.4722	251.5111	129.6326	217.835
0.23	2662.482	902.0254	282.2544	136.8255	212.5315
0.240000	3008.532	1022.021	316.5027	147.0727	206.7553
0.25	3384.255	1145.724	353.3447	154.3564	203.5441
0.26	3789.952	1281.672	393.2257	164.7028	202.124
0.27	4222.212	1422.132	436.2577	176.1203	201.4558
0.28	[illegible]	1571.548	482.77	[illegible]	200.88[illegible]
0.29	5187.487	1730.332	532.242	202.3120	201.3038
0.3	5721.607	1898.742	585.4125	217.1157	201.7511
0.31	6287.022	2080.213	642.1722	233.0513	201.3757
0.32	[illegible]	2272.391	705.5575	250.2632	203.5132
0.33	7524.523	2451.255	768.9814	268.6751	205.5701
0.34	8202.787	2723.501	835.2045	288.2279	208.5401
0.349999	8970.621	2925.616	907.5001	309.2368	212.5134
0.359999	9746.727	3137.495	982.7521	331.[illegible]	216.6805
0.369999	10560.37	3517.722	1064.704	355.1625	221.4614
0.379999	[illegible]	[illegible]	1149.041	380.[illegible]	226.9159
0.389999	[illegible]	4167.222	1237.[illegible]	406.2483	233.7201
0.399999	[illegible]	4447.902	1330.722	434.3247	[illegible].2372
0.409999	14255.21	4811.226	1423.728	462.856	245.4774
0.419999	15300.37	5173.432	1522.41	494.5057	254.2127
0.429999	16382.38	5546.738	1645.421	525.86	261.5268
0.439999	17522.59	5932.237	1758.331	560.7795	271.5016
0.449999	[illegible]771.71	6331.321	1875.427	596.3010	[illegible]
0.459999	[illegible]	[illegible]744.254	[illegible]	637.4507	[illegible]
0.469999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.479999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.489999	24024.[illegible]	[illegible]	2400.[illegible]	755.0745	335.4425
0.499999					

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Level [illegible]

0.509999	25527.23	8823.924	2585.843	799.107	338.1191
0.519999	27053.53	9151.851	2708.517	844.9414	351.4292
0.529999	28549.73	9589.649	2887.189	892.6123	365.4117
0.539999	30302.13	10248.34	3031.975	942.1336	380.0448
0.549999	32013.89	10822.12	3202.928	993.5905	395.2449
0.559999	33755.10	11423.37	3380.348	1045.983	411.322
0.569999	35522.02	12072.48	3554.181	1102.339	427.9641
0.579999	37246.72	12637.22	3724.549	1159.702	445.2227
0.589999	39732.43	13385.85	3901.527	1219.105	460.3877
0.599999	[illegible]	[illegible]	[illegible]	1280.583	482.1271
0.609999	[illegible]	[illegible]	[illegible]	1344.171	[illegible]
0.619999	42852.80	15505.63	4524.136	1409.902	521.8772
0.629999	48117.90	16272.16	4808.115	1477.811	542.0255
0.639999	50445.76	17025.20	5041.26	1547.932	564.0449
0.649999	52847.5	[illegible]	5280.99	1620.201	587.0119
0.659999	55024.87	[illegible]	5523.052	1694.921	610.2471
0.669999	[illegible]	[illegible]	[illegible]	1771.917	[illegible]
0.679999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.689999	[illegible]	[illegible]	[illegible]	1932.924	[illegible]
0.699999	[illegible]	22220.22	6592.797	2017.024	711.0699
0.709999	[illegible]	[illegible]	[illegible]	[illegible]	738.2099
0.719999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.729999	[illegible]	[illegible]	[illegible]	2284.878	795.2452
0.739999	77573.82	26063.52	7768.804	2376.423	824.989
0.749999	81162	27402.24	8106.342	2475.161	855.5951
0.759999	84472.79	[illegible]	8428.783	2574.522	887.0684
0.769999	[illegible]	[illegible]	8772.87	2675.552	919.4191
0.779999	91316.69	30673.82	9119.891	2781.278	952.664
0.789999	94377.15	[illegible]	9474.901	2888.721	986.8107
0.799999	[illegible]	[illegible]	9823.017	2998.502	1021.87
0.809999	102227	34221.45	10212.35	3111.974	1057.822
0.819999	105101.5	35378.04	10595.02	3227.832	1094.788
0.829999	[illegible]	37205.52	[illegible]	3345.55	1132.622
0.839999	[illegible]	38227.64	[illegible]	3465.192	1171.442
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.859999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.869999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.879999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.889999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.899999	140222.5	47428.05	14006.32	4250.924	1424.995
0.909999	[illegible]	[illegible]	[illegible]	4403.898	1470.8
0.919999	[illegible]	[illegible]	[illegible]	[illegible]	1517.546
0.929999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.939995	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.979999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.989999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.999999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.009999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.019999	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

Level Flight [illegible]

1.065999	237727.8	78712.25	23332	7147.531	2332.[illegible]
1.073999	242409.4	81955.13	24138.25	7349.487	2417.822
1.089999	249205.5	84255.14	24877.15	7555.071	2484.075
1.099999	25[illegible]27.4	80808.88	25257.59	7764.464	2551.88[illegible]

9.2.2 Turning Flight

```
100 REM           Flight Preformance
110 REM
120 REM           Turning Flight
130 REM
140 REM ****************         Variables and Constants          ***************
150 REM
160 REM           M  - Mach Number
170 REM           S  - Wing Area  (ft^2)
180 REM          V  - Velocity  (ft/s)
190 REM          AR - Aspect Ratio
200 REM          FE - Bank Angle  (degrees)
210 REM          NF - Bank angle (radians)
220 REM          L  - Lift  (lb)
230 REM          W  - Weight  (lb)
240 REM          CD - Coefficient of Drag
250 REM          CO - Coefficient of Drag at zero lift
260 REM          K  - Constant
270 REM          CL - Coefficient of Lift
280 REM          PI - 3.14
290 REM          P  - Power  (hp)
300 REM          D  - Drag
310 REM          E  - Airplane efficiency
320 REM          PA - Power available (can be generated by the motor)
330 REM          RG - Power generated/power received by motor
340 REM          EM - Motor efficiency
350 REM          EP - Propeller efficiency
360 REM          PD - Power Density (hp/ft^2)
370 REM          FA - Altitude (ft)
380 REM          PR - Power required (must be supplied to the motor)
390 REM          T  - Thrust
400 REM          FR - Flight radius
410 REM          AI - Angle of incidence
420 REM          AD - Angle of incidence (degrees)
430 REM
440 REM ****************         Define Constants         *************************
450 REM
460              E  = .85            :REM  Airolane efficiency
470              PI = 3.1416         :REM  Value of one radian
480              I  = .01            :REM  Increase the Mach #
490              ST = 2              :REM  Increase the bank angle
500              K  = .007           :REM  Viscous Drag Coefficient estimated
                                           from a NACA 2214 airfoil
510              G  = 32.174         :REM  Gravitational constant  (ft/s^2)
520              RG = .97            :REM  Power recieved/power generated
530              EM = .98            :REM  Motor efficiency
540              AP = 175             :REM  Propeller area  (ft^2)
550              CP = 550            :REM  Conversion from hp to ft*lbf/s
560 REM
570              PD = (700*1.3405*10^-3*(.3048)^2)
580 REM
590 REM ****************        Data Entry        **********************************
600 REM
610              INPUT "Gross Weight";W
620              INPUT "Wing Area"; S
630              INPUT "Aspect Ratio";AR
640              INPUT "Altitude of flight";FA
650              INPUT "Speed of sound at altitude";SS
660              INPUT "Density at altitude";RO
670              INPUT "Coefficient of Drag @ zero lift";CO
680              INPUT "Total propeller area";AP
690 REM
700 REM ****************          Print Input Constants    *************************
```

```
720          LPRINT "Gross Weight                              ";W
730          LPRINT "Wing Area                                 ";S
740          LPRINT "Aspect Ratio                              ";AR
750          LPRINT "Altitude of flight                        ";FA
760          LPRINT "Speed of sound @ altitude                 ";SS
770          LPRINT "Density at altitude                       ";RO
780          LPRINT "Coefficient of Drag @ zero lift           ";CO
             LPRINT "Total propeller area                      ";AP
800 REM
810 REM ****************          Calculations     ******************************
820 REM
830          OPEN "c:50CR.out" FOR OUTPUT AS 1 LEN=2000
840 REM
850          FOR FE = 3 TO 13 STEP ST
860 REM
870                  NF = FE*PI/180
880                  L = (W/COS(NF))
890 REM
900                  PRINT "Bank Angle   ";FE;  "degrees"
910                  PRINT#1, "Bank Angle";FE
920                  PRINT
930                  PRINT "Mach #","Vel","CL","Climb Rate"
940                  PRINT
950 REM
960                  FOR M = .01 TO 1.1 STEP I
970 REM
980                          V = M*SS
990 REM
1000                         CL = (L/(.5*RO*S*(V^2)))
1010                         CD = (CO + K*(CL^2) + (CL^2)/(E*PI*AR) )
1020                         D = CD*.5*RO*(V^2)*S
1030 REM
1040                         FR = (V^2/(G*TAN(NF)))
1050                         AI = ATN(V^2/(FR*G)) + ATN(FR/FA)
1060                         AD = AI*(180/PI)
1070 REM
1080                         P = D*V/CP           :REM Power Required  (hp)
1090                         PR = PD*S*COS(AI)    :REM Power Received
1100 REM
1110                         T = ((PR/V)*CP)
1120 REM
1130                         EP = (2/(1+(1+(T/(.5*RO*(V)^2*AP)))^.5))
1140 REM
1150                         PA = RG*EM*EP*PR     :REM Power Available
1160 REM
1170                         DH = ((PA - P)/W)*CP
1180 REM
1190                         PRINT M,V,CL,PA,PR,DH
1200                         PRINT#1, M;V;CL;PA;P;DH
1210 REM
1220                 NEXT M
1230 REM
1240         NEXT FE
1250 REM
1260         CLOSE 1
1270 REM
1280         END
```

Turning Flight Analysis

Altitude Sea Level to 100,00 feet

Power Available (Bank Angle = 15)

Mach #	Altitude (feet) Sea Lev	25,000	50,000	75,000	100,000
0.01		12.59245	7.221528	4.15553	2.27332
0.02		27.05194	20.75065	11.62576	6.27715
0.03		54.22441	21.92255	21.00123	12.14706
0.04		92.2853	54.82140	31.85322	17.46054
0.05		118.9028	73.5746	42.35522	25.49553
0.06		145.2004	92.23256	55.44215	33.02377
0.07		167.7240	110.7552	67.55777	40.92475
0.08		187.1889	128.43	80.59592	49.20182
0.089999		203.2853	145.008	93.19324	57.85124
0.099999		217.1821	160.2432	105.59	66.22744
0.11		228.0827	174.3512	117.6585	74.5911
0.12		235.3361	186.3757	129.2204	82.84052
0.13		242.8517	195.2277	140.455	91.15122
0.14		248.9216	206.183	151.1127	100.6759
0.15		252.8452	216.3051	161.1242	109.024
0.16		[illegible]	224.[illegible]	[illegible]	[illegible]
0.17		[illegible]	[illegible]	177.2322	125.5225
0.18		252.5372	236.8058	187.4247	133.02
0.19		253.2548	241.2522	194.5227	140.5285
0.2		255.6455	245.3454	201.8043	147.7522
0.21		257.5715	248.5530	208.0915	154.7065
0.22		255.8153	251.3525	213.8067	161.3494
0.23		254.9245	253.5921	218.572	167.5829
0.240000		252.5042	255.3155	222.8243	173.7013
0.25		250.2732	255.522	227.2055	179.4018
0.26		247.2864	257.5487	231.5215	184.7827
0.27		244.4248	258.1222	234.212	189.5456
0.28		240.5978	258.4084	237.7053	194.5251
0.29		237.274	258.4[illegible]	240.5[illegible]	[illegible]
0.3		232.2511	258.1245	242.4027	202.1777
0.31		229.0252	257.1194	244.2142	207.0152
0.32		224.2552	256.2272	245.5227	210.5701
0.33		219.2579	255.3224	247.1047	213.2402
0.34		214.9811	254.5255	248.1264	216.8285
0.349999		209.919	252.5117	248.9055	219.5731
0.359999		204.7018	252.0182	249.4521	222.0522
0.369999		[illegible]	250.2575	249.7572	224.2125
0.379999		192.2824	248.5275	249.9277	226.2275
0.389999		[illegible]	246.255	249.2052	228.0252
0.399999		181.6704	244.448	249.8521	229.[illegible]
0.409999		177.[illegible]	242.1557	249.5524	230.557
0.419999		171.2022	239.7558	248.7144	232.0505
0.429999		165.5734	237.27	248.0052	233.0452
0.439999		159.9224	234.624	247.152	233.5119
0.449999		154.1515	231.65	246.1572	234.2555
0.459999		[illegible]	228.5506	245.047	234.5102
0.469999		[illegible]	[illegible]	243.5015	235.[illegible]
0.479999		[illegible]	[illegible]	[illegible]	[illegible]
0.489999		[illegible]	[illegible]	[illegible]	[illegible]
0.499999		[illegible]	212.4711	233.2043	234.2522

0.509999	121.1411	212.1274	237.8887	234.328
0.519999	115.9333	209.653	235.2471	234.0528
0.529999	110.845	205.1952	233.9434	233.4413
0.539999	103.8819	202.6332	231.945	232.704
0.549999	100.5944	199.0135	229.8551	231.8454
0.559999	96.2455	195.3425	227.6306	230.8731
0.569999	91.5174	191.5255	225.4222	229.7535
0.579999	87.11147	187.5722	222.022	228.537
0.589999	82.7214	184.0255	220.3737	227.2027
0.599999	78.4221	180.2738	218.1871	225.5055
0.609999	74.22004	176.4404	215.5222	224.42?7
0.619999	70.31324	172.8939	213.0226	222.842
0.629999	66.41223	168.7292	210.3494	221.1772
0.639999	62.62722	164.8218	207.6194	219.4282
0.649999	58.97488	161.0271	204.8275	217.5892
0.659999	55.42228	157.1901	202.0072	215.652
0.669999	51.93722	152.3752	199.1227	213.7127
0.679999	48.82772	149.2237	196.3188	211.5722
0.689999	45.42257	145.7272	193.2272	209.3822
0.699999	41.31255	141.5222	190.2829	207.22
0.709999	38.29.?	138.2217	187.2737	205.1252
0.719999	[illegible]	[illegible]	[illegible]	[illegible]
0.729999	[illegible]	[illegible]	[illegible]	[illegible]
0.739999	27.72238	127.2106	176.0042	198.122
0.749999	28.1482	122.8194	172.0019	195.7256
0.759999	27.22377	120.0722	171.2881	193.2872
0.769999	22.09981	116.2718	166.7352	190.7493
0.779999	20.32514	113.1182	162.6898	188.2079
0.789999	19.29745	109.7149	161.2817	185.8245
0.799999	16.11594	106.2827	159.4248	183.0223
0.809999	13.92550	103.0827	155.2225	180.4028
0.819999	11.52291	99.21614	150.2153	177.7252
0.829999	9.777488	95.52253	150.121	175.0279
0.839999	7.797127	92.48258	147.0272	172.4028
0.849999	[illegible]	[illegible]	[illegible]	[illegible]
0.859999	[illegible]	[illegible]	[illegible]	[illegible]
0.869999	[illegible]	[illegible]	[illegible]	[illegible]
0.879999	[illegible]	[illegible]	[illegible]	[illegible]
0.889999	[illegible]	74.92271	131.5775	158.2222
0.899999	-2.86822	75.82248	130.9113	156.0927
0.909999	-4.48122	73.0754	125.5705	153.3527
0.919999	-2.00942	70.28218	122.0271	150.841
0.929999	[illegible]	67.7422	120.1724	147.8212
0.939999	-1.38688	62.12840	117.3122	145.2054
0.949999	[illegible]	[illegible]	[illegible]	[illegible]
0.959999	[illegible]	60.14267	[illegible]	[illegible]
0.969999	[illegible]	[illegible]	[illegible]	[illegible]
0.979999	[illegible]	62.22372	106.3156	124.4632
0.989999	[illegible]	52.01257	103.319	121.3152
0.999999	-11.8448	50.73238	100.0721	128.1257
1.009999	-9.1309	48.20735	98.22271	125.2714
1.019999	[illegible]	[illegible]	[illegible]	[illegible]
1.029999	[illegible]	[illegible]	[illegible]	[illegible]
1.039999	[illegible]	[illegible]	[illegible]	[illegible]
1.049999	[illegible]	[illegible]	[illegible]	[illegible]
1.059999	-22.4887	[illegible]	[illegible]	[illegible]

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1.059995	-24.4916	35.11455	83.25272	111.3231
1.073959	-25.4578	34.20225	80.52802	108.8762
1.089994	-26.4185	32.33373	78.60019	106.4453
1.099993	-27.3444	30.50451	76.30917	104.0405

Turning Flight Analysis

Altitude Sea Level to 100,00 feet

Mach #	Sea Lev	25,000	50,000	75,000	100,000
	Power Required (Bank Angle = 15)	Altitude (feet)			
0.01		142.9135	439.3352	1448.737	4803.904
0.02		71.54432	219.8121	724.412	2401.965
0.03		49.37303	145.5378	483.0675	1601.351
0.04		39.87549	111.0255	362.5554	1201.095
0.05		32.70233	90.25425	290.4748	961.0151
0.06		27.8628	77.38619	242.7132	801.0578
0.07		42.72529	69.34752	208.9593	685.9187
0.08		51.17054	64.74854	184.0734	601.4536
0.089999		52.3067	62.5148	165.2161	535.142
0.099999		73.22294	62.13855	150.6975	482.2765
0.11		99.29185	65.50232	139.455	439.2293
0.12		124.3417	69.79919	130.752	403.3549
0.13		152.543	75.59105	124.2359	373.6734
0.14		182.7503	84.08338	119.4828	348.3121
0.15		229.1281	94.11092	116.2392	326.5258
0.16		270.427	[illegible]	114.4032	307.9708
0.17		322.0823	121.2058	113.8344	291.8504
0.18		387.4133	136.4309	114.4026	277.885
0.19		453.8208	154.8518	116.1981	265.7758
0.2		527.6875	175.5207	119.0212	255.2862
0.21		609.3933	198.7948	122.9254	245.2273
0.22		699.3387	224.4837	127.869	238.4457
0.23		797.855	252.7906	133.853	231.8173
0.240000		905.454	283.8205	140.8754	226.2403
0.25		1022.404	317.6799	148.9545	221.6213
0.26		1149.132	354.4785	158.065	217.9211
0.27		1285.025	394.3155	168.2071	215.0825
0.28		1423.477	437.3188	179.3962	212.5785
0.29		1591.372	483.5242	192.005	211.8535
0.3		1751.8	533.223	205.6481	211.0628
0.31		1940.051	586.3859	220.3461	211.1822
0.32		2129.615	643.0591	236.1247	211.922
0.33		2342.53	703.349	253.2101	213.2549
0.34		2561.637	767.8269	271.5294	215.4349
0.349999		2793.674	835.0455	291.1107	218.1244
0.359999		3039.784	908.2175	311.9628	221.4543
0.369999		3299.754	984.7574	334.1748	225.3378
0.379999		3574.174	1065.479	357.7173	229.9259
0.389999		3863.435	1150.595	382.5408	235.0772
0.399999		4167.328	1240.221	408.5753	240.8251
0.409999		4486.04	1334.489	435.7537	247.1819
0.419999		4824.163	1433.456	465.0058	254.1254
0.429999		5175.563	1537.295	495.7555	261.7019
0.439999		5545.004	1646.1	529.0655	269.8719
0.449999		5932.5	1759.955	563.5352	278.652
0.459999		6335.597	1878.088	[illegible]	288.0427
0.469999		6755.593	[illegible]	[illegible]	298.0529
0.479999		[illegible]	[illegible]	[illegible]	[illegible]
0.489999		[illegible]	[illegible]	[illegible]	[illegible]
0.499999		[illegible]	[illegible]	757.0152	331.8755

0.509999	8634.141	2556.42	801.0095	344.425
0.519999	9151.836	2709.083	846.8076	357.6277
0.529999	9589.829	2857.745	894.4435	371.4834
0.539999	10248.52	3032.52	943.9509	386.0035
0.549999	10822.29	3203.523	995.3523	401.1955
0.559999	11425.54	3380.871	1048.716	417.0585
0.569999	12052.55	3564.577	1104.041	433.6297
0.579999	12698.02	3755.027	1161.375	450.882
0.589999	13366.02	3952.155	1220.75	468.855
0.599999	14057.08	4155.357	1282.201	487.5305
0.609999	14771.55	4365.735	1345.782	505.9227
0.619999	15509.84	4584.81	1411.468	527.0675
0.629999	16272.22	4809.583	1479.352	547.9445
0.639999	17055.42	5041.819	1549.45	569.573
0.649999	17871.5	5281.432	1621.794	591.9525
0.659999	18708.92	5528.541	1696.421	615.122
0.669999	19572.19	5782.229	1773.385	639.0537
0.679999	20461.57	6045.701	1852.62	663.7345
0.689999	21377.5	6315.323	1934.241	689.3255
0.699999	22320.37	6594.217	2018.441	715.5671
0.709999	23290.57	6880.722	2104.995	742.6286
0.719999	24282.42	7173.[illegible]	2194.74	770.5005
0.729999	25314.51	7477.795	2285.505	799.5222
0.739999	26369.05	7789	2379.755	829.3272
0.749999	27452.47	8108.725	2476.454	859.8826
0.759999	28565.17	8437.113	2575.802	891.2994
0.769999	29707.55	8774.552	2677.812	923.5582
0.779999	30879.98	9120.269	2782.519	956.7895
0.789999	32082.87	9475.274	2889.955	990.8842
0.799999	33316.8	9839.363	3000.169	1025.852
0.809999	34581.57	10212.72	3113.172	1061.825
0.819999	35878.16	10595.39	3229.015	1098.692
0.829999	37206.75	10987.5	3347.726	1136.505
0.839999	38567.76	11389.19	3469.348	1175.274
0.849999	39961.55	11800.95	3593.907	1215.01
0.859999	41388.85	12221.72	3721.442	1255.724
0.869999	42849.11	12652.2	3851.989	1297.425
0.879999	44342.58	13092.9	3985.578	1340.125
0.889999	45872.52	13542.12	4122.247	1383.835
0.899999	47436.15	14006.65	4262.032	1428.572
0.909999	49034.93	14478.53	4404.954	1474.335
0.919999	50669.21	14960.69	4551.08	1521.144
0.929999	52339.43	15451.85	4700.415	1569.004
0.939999	54045.94	15957.55	4853.004	1617.93
0.949999	55789.18	16472.07	5008.881	1667.931
0.959999	57569.42	16997.54	5168.082	1719.018
0.969999	59387.58	17524.07	5330.641	1771.203
0.979999	61242.91	18081.79	5496.592	1824.495
0.989999	63135.85	18640.3	5665.972	1878.909
0.999999	65069.41	19211.21	5838.814	1934.452
1.009999	67041.01	19793.15	6015.152	1991.135
1.019999	69052.04	20388.72	6195.020	2048.972
1.029999	71102.51	20991.07	6378.455	2107.974
1.039999	73192.57	21607.02	6565.304	2168.147
1.049999	75322.55	22237.47	6755.301	2229.51
1.059999	77492.34	22878.75	6950.500	2292.087

1.069999	79712.38	23333.28	7148.588	2355.632
1.079999	81980.22	24199.12	7350.386	2420.817
1.089999	84265.22	24877.41	7555.952	2487.031
1.099999	86506.73	25568.26	7763.346	2554.489

9.3 Climb Performance

9.3.1 Climb Rate

Altitude Sea Level to 100,000 feet

Climb Rate (Bank Angle = 10)

Altitude (feet)

Mach #	Sea Lev	25,000	50,000	75,000	100,000
0.01		-10.1231	-23.5775	-113.748	-378.128
0.02		-2.80222	-15.5540	-55.0827	-186.811
0.03		1.425258	-8.21272	-35.3072	-127.108
0.04		4.508610	-4.20575	-25.9234	-93.0886
0.05		7.019592	-1.02505	-18.8590	-73.5756
0.06		9.01281	1.544112	-14.2209	-60.3556
0.07		10.43104	3.555905	-10.8381	-50.7106
0.08		11.40294	5.51444	-7.83327	-43.2931
0.089999		11.7715	7.022227	-5.30750	-37.3723
0.099999		11.27567	8.288444	-3.14009	-32.5082
0.11		10.52229	9.229142	-1.22791	-28.3895
0.12		9.517724	9.919412	0.288420	-24.8772
0.13		7.848319	10.34255	1.822535	-21.8105
0.14		5.80889	10.50132	3.071319	-19.1095
0.15		3.195272	10.39912	4.147192	-16.7103
[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
[illegible]		-3.51075	[illegible]	5.815242	-12.8257
0.18		-10.4418	8.546282	6.423247	-10.8931
0.19		-15.9194	7.417838	6.895416	-9.32772
0.2		-22.0559	6.024114	7.221724	-7.90827
0.21		-28.9145	4.382844	7.411512	-6.62730
0.22		-36.4839	2.494173	7.457605	-5.47285
0.23		-44.8042	0.332281	7.3924	-4.42502
0.240000		-53.9032	-2.09533	7.187319	-3.50991
0.25		-63.8095	-4.79414	6.855844	-2.68818
0.26		-74.5513	-7.78929	6.397518	-1.98591
0.27		-86.1570	-11.0857	5.812958	-1.33779
0.28		-98.6502	-14.5729	5.105514	-0.80092
0.29		[illegible]	[illegible]	[illegible]	[illegible]
0.3		-126.443	-22.8834	3.317648	[illegible]
0.31		-141.789	-27.0142	2.217918	0.284755
0.32		-158.140	-31.7270	1.022702	0.459725
0.33		-175.520	-36.8355	-0.28722	[illegible]
0.34		-193.982	-42.3179	-1.74995	0.667830
0.349999		-213.528	-48.0582	-3.32109	0.640181
0.359999		-234.182	-54.8165	-5.02982	[illegible]
0.369999		[illegible]	[illegible]	-6.87748	[illegible]
0.379999		[illegible]	-67.8485	-8.84827	[illegible]
0.389999		[illegible]	-74.7727	[illegible]	[illegible]
0.399999		[illegible]	-82.3801	[illegible]	-0.58108
0.409999		[illegible]	-90.2750	-15.5427	-1.03247
0.419999		-382.4	[illegible]	-18.0504	-1.55822
0.429999		-412.751	-107.584	-20.5937	-2.14910
0.439999		-442.484	-116.808	-23.4775	-2.80728
0.449999		-475.572	-126.455	-26.4037	-3.52273
0.459999		[illegible]	[illegible]	-29.4742	[illegible]
[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
[illegible]		-622.123	[illegible]	-42.8402	-8.18761

[illegible]

0.509999	-699.229	-192.728	-47.0888	-9.28117
0.519999	-742.011	-206.592	-51.0421	-10.4707
0.529999	-785.411	-219.948	-55.1800	-11.7252
0.539999	-832.492	-233.791	-59.4788	-13.0477
0.549999	-880.287	-248.120	-63.9404	-14.4353
0.559999	-929.827	-262.928	-68.5672	-15.8891
0.569999	-981.148	-278.432	-73.3612	-17.4081
0.579999	-1034.27	-294.372	-78.3246	-18.9984
0.589999	-1089.24	-310.828	-83.4588	-20.6463
0.599999	-1145.08	-327.892	-88.7683	-22.3676
0.609999	-1204.94	-345.492	-94.2571	-24.1225
0.619999	-1265.82	-363.684	-99.9160	-26.0070
0.629999	-1328.19	-382.415	-105.759	-27.9273
0.639999	-1392.68	-401.757	-111.785	-29.9191
0.649999	-1459.55	-421.597	-117.997	-31.9705
0.659999	-1528.34	-442.244	-124.398	-34.0940
0.669999	-1599.31	-463.405	-130.984	-36.2888
0.679999	-1672.33	-485.200	-137.765	-38.5430
0.689999	-1747.29	-507.627	-144.740	-40.8743
0.699999	-1824.75	-530.698	-151.910	-43.2730
0.709999	-1904.28	-554.424	-159.280	-45.7407
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
0.729999	-2070.09	-604.878	-174.627	-50.8969
0.739999	-2158.83	-629.881	-182.682	-53.5860
0.749999	-2245.72	-658.088	-190.819	-56.3184
0.759999	-2336.02	-682.157	-199.229	-59.1265
0.769999	-2430.88	-711.047	-207.849	-62.0221
0.779999	-2526.74	-739.617	-216.689	-65.0002
0.789999	-2625.34	-768.917	-225.749	-68.0413
0.799999	-2725.47	-798.956	-235.032	-71.1962
0.809999	-2830.18	-829.745	-244.541	-74.3455
0.819999	-2936.41	-861.292	-254.279	-77.6102
0.829999	-3045.39	-893.605	-264.247	-80.9507
0.839999	-3156.81	-926.699	-274.45	-84.3679
0.849999	-3271.08	-960.575	-284.885	-87.8581
0.859999	-3387.94	-995.256	-295.567	-91.4348
0.869999	-3507.61	-1030.73	-306.488	-95.0851
0.879999	-3630.08	-1067.02	-317.654	-98.8123
0.889999	-3755.31	-1104.16	-329.068	-102.622
0.899999	-3883.40	-1142.12	-340.732	-106.517
0.909999	-4014.27	-1180.92	-352.651	-110.49
0.919999	-4148.25	-1220.58	-364.825	-114.549
0.929999	-4285.05	-1261.10	-377.259	-118.684
0.939999	-4424.84	-1302.49	-389.955	-122.907
0.949999	-4567.62	-1344.77	-402.916	-127.215
0.959999	-4713.44	-1387.94	-416.147	-131.608
0.969999	-4862.32	-1432.01	-429.648	-136.089
0.979999	-5014.31	-1476.99	-443.422	-140.657
0.989999	-5169.41	-1522.89	-457.473	-145.313
0.999999	-5327.69	-1569.72	-471.804	-150.059
1.009999	-5489.15	-1617.50	-486.417	-154.895
1.019999	-5653.85	-1666.22	-501.318	-159.822
1.029999	-5821.79	-1715.90	-516.508	-164.841
[illegible]	[illegible]	-1766.[illegible]	[illegible]	-169.[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.059999	-6342.51	-1870.20	-580.824	-180.428

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[illegible]

1.069999	-6528.38	-1924.41	-500.194	-185.[illegible]
1.079999	-6711.55	-1978.02	-555.257	-191.249
1.089999	-6899.72	-2024.57	-513.845	-195.925
1.099999	-7091.27	-2091.33	-831.123	-202.929

9.4 Flight Path

50,000 feet

Altitude 50,000 feet

Mach #	Velocity	C Lift	P Avail	P Req	Cl Rate
Bank Ang	3				
0.01	9.8802	103.5571	7.549754	411.0333	-23.0220
0.02	19.3505	27.14575	21.13117	205.6507	-15.1080
0.03	29.0409	12.05523	37.64881	137.5233	-8.17599
0.04	38.7212	6.788851	55.25328	102.9828	-3.93531
0.05	48.4015	4.343482	74.37342	84.60357	-0.78225
0.06	58.0818	3.016307	94.18496	72.65102	1.782817
0.07	67.7621	2.216063	112.5975	65.30486	3.904296
0.08	77.44239	1.695673	130.9981	61.21056	5.712989
0.089999	87.12259	1.340081	147.8759	59.67002	7.220857
0.099999	96.80199	1.085871	163.4119	60.30835	8.440345
0.11	106.4822	0.897413	177.4549	62.92024	9.378032
0.12	116.1626	0.754078	189.9191	67.44091	10.03842
0.13	125.8429	0.642527	200.7858	73.81291	10.35363
0.14	135.5242	0.554012	209.9745	82.05174	10.47293
0.15	145.2045	0.482609	217.5225	92.22455	10.28527
0.16	154.8848	0.424153	222.7232	104.3897	9.771321
0.17	164.5651	0.375732	226.3437	118.5408	8.985764
0.18	174.2454	0.335145	231.5525	134.8485	7.916455
0.19	183.9257	0.300795	232.4363	153.3722	6.554269
0.2	193.606	0.271457	234.0325	174.2055	4.901594
0.21	203.2863	0.246225	233.5847	197.4468	2.958348
0.22	212.9666	0.224353	232.0408	223.1971	0.723969
0.23	222.647	0.205265	229.5528	251.5601	-1.80157
0.240000	232.3273	0.188519	226.2252	282.6412	-4.61927
0.25	242.0076	0.173739	222.164	316.5478	-7.72551
0.26	251.6879	0.160621	217.4749	353.388	-11.1252
0.27	261.3681	0.148923	212.2617	393.2712	-14.8175
0.28	271.0484	0.138503	206.554	436.3079	-18.8025
0.29	280.7287	0.129118	200.5282	482.6086	-23.0814
0.3	290.409	0.120653	194.4453	532.2855	-27.6554
0.31	300.0893	0.112993	188.0721	585.4438	-32.5304
0.32	309.7696	0.106042	181.5083	642.2141	-37.7066
0.33	319.4499	0.099712	175.1103	702.6914	-43.1892
0.34	329.1302	0.093933	158.6375	766.9946	-48.9831
0.349999	338.8105	0.088642	162.233	835.2368	-55.0539
0.359999	348.4907	0.083785	155.5234	907.5214	-61.5270
0.369999	358.171	0.079318	149.7655	983.9925	-68.2916
0.379999	367.8513	0.075199	143.7615	1064.724	-75.3533
0.389999	377.5316	0.071391	137.5301	1149.865	-82.8401
0.399999	387.2119	0.067866	132.2864	1239.513	-90.5405
0.409999	396.8922	0.064596	126.8369	1333.779	-98.8034
0.419999	406.5725	0.061557	121.5922	1432.782	-107.227
0.429999	416.2528	0.058727	118.6485	1535.636	-116.258
0.439999	425.9331	0.056086	111.7074	1645.455	-125.555
0.449999	435.6134	0.053623	107.0564	1759.325	-135.257
0.459999	445.2937	0.051317	102.3218	1878.481	-145.374
0.469999	454.974	0.049155	98.08327	2002.855	-155.775
0.479999	464.6543	0.047125	94.20124	2132.585	-166.357
0.489999	474.3346	0.045221	91.41228	2268.053	-173.287

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0.499999	484.0143	0.043434	86.59861	2409.074	-190.116
0.509999	493.8312	0.041748	83.14951	2555.855	-202.422
0.519999	503.2755	0.040157	79.75938	2708.539	-215.199
0.529999	513.0857	0.038656	76.52119	2867.511	-228.453
0.539999	522.7351	0.037238	73.42792	3031.995	-242.155
0.549999	532.4153	0.035896	70.47276	3202.009	-256.438
0.559999	542.0858	0.034623	67.64916	3380.268	-271.188
0.569999	551.7793	0.033421	64.95083	3554.181	-286.455
0.579999	561.4571	0.032279	62.37109	3734.559	-302.253
0.589999	571.1372	0.031194	59.90447	3951.842	-318.582
0.599999	580.8173	0.030163	57.54512	4155.025	-335.471
0.609999	590.4961	0.029182	55.28771	4366.323	-352.913
0.619999	600.1734	0.028248	53.12705	4584.154	-370.922
0.629999	609.8525	0.027358	51.05812	4809.134	-389.509
0.639999	619.535	0.026510	49.07526	5041.378	-408.683
0.649999	629.2191	0.025701	47.17727	5280.997	-428.454
0.659999	638.8993	0.024928	45.35890	5528.113	-448.823
0.669999	648.5711	0.024188	43.61104	5782.207	-469.[illegible]
0.679999	658.2601	0.023483	41.93504	6045.394	-491.459
0.689999	667.9404	0.022807	40.32831	6315.572	-512.706
0.699999	677.6207	0.022160	38.7345	6593.813	-535.612
0.709999	[illegible]	0.021540	[illegible]	[illegible]	-558.[illegible]
0.719999	696.9811	0.020948	35.8796	7174.613	-584.392
0.729999	706.6612	0.020375	34.5064	7477.41	-609.298
0.739999	716.2418	0.019825	33.18852	7788.519	-634.880
0.749999	725.9221	0.019304	31.92062	8108.356	-661.159
0.759999	735.7124	0.018799	30.70011	8436.741	-688.141
0.769999	745.3827	0.018314	29.52476	8773.887	-715.837
0.779999	755.083	0.017847	28.39246	9119.905	-744.256
0.789999	764.7433	0.017399	27.30131	9474.914	-773.407
0.799999	774.4138	0.016966	26.24923	9839.029	-803.301
0.809999	784.1033	0.016550	25.23453	10212.36	-833.946
0.819999	793.7843	0.016149	24.25658	10595.04	-865.353
0.829999	803.4645	0.015762	23.31060	10987.18	-897.530
0.839999	813.1447	0.015389	22.3987	11388.85	-930.488
0.849999	822.822	0.015029	21.51702	11800.23	-964.237
0.859999	832.[illegible]	0.014681	20.66847	12221.39	-998.784
0.869999	842.1[illegible]	0.014346	19.84829	12652.47	-1034.14
0.879999	851.8[illegible]	0.014023	19.04534	13093.58	-1070.31
0.889999	861.5462	0.013708	18.2757	13544.83	-1107.32
0.899999	871.2265	0.013405	17.53185	14006.34	-1145.16
0.909999	880.9068	0.013112	16.8109	14478.22	-1183.85
0.919999	890.5871	0.012829	16.11285	14960.59	-1223.39
0.929999	900.2673	0.012554	15.43681	15453.55	-1263.80
0.939999	909.9475	0.012289	14.7818	15957.25	-1305.09
0.949999	919.6271	0.012031	14.14704	16471.77	-1347.25
0.959999	929.3082	0.011782	13.53158	16997.85	-1390.32
0.969999	938.9885	0.011540	12.93492	17533.78	-1434.30
0.979999	948.6688	0.011306	12.35612	18081.5	-1479.18
0.989999	958.3491	0.011079	11.79447	18640.51	-1524.99
0.999999	968.0294	0.010858	11.24925	19210.92	-1571.73
1.009999	977.7096	0.010644	10.72017	19792.87	-1619.42
1.019999	987.3899	0.010437	10.20625	20386.45	-1668.[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	20991.75	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	22223.2	-181[illegible].79

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1.049999	1065.111	0.009554	8.292427	22973.5	-1872.3
1.059999	1025.791	0.009484	7.846809	23533.01	-1925.83
1.079999	1045.472	0.009309	7.413409	24198.66	-1930.37
1.089999	1053.152	0.009139	6.991798	24877.15	-2022.93
1.099999	1054.822	0.008974	6.591508	25568.01	-2092.52

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Bank Ang 5

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0.01	9.6203	108.2224	7.940319	413.0448	-33.1675
0.02	19.2876	27.21311	21.10452	206.6024	-15.1908
0.03	29.0409	12.09470	37.80048	129.1908	-8.22424
0.04	38.7216	6.803276	55.62622	104.4858	-3.58325
0.05	48.4015	4.354092	74.87932	82.00599	-0.82999
0.06	58.0918	3.02269	94.07729	72.53523	1.725759
0.07	67.7821	2.221479	113.6926	65.79162	3.872271
0.08	77.44237	1.70082	130.969	61.4621	5.595747
0.089999	87.12269	1.343656	147.891	59.08252	7.203712
0.099999	96.80199	1.088225	162.5732	[illegible]	8.437223
0.11	106.4802	0.899507	177.8713	56.1121	9.294013
0.12	116.1621	0.755916	190.7097	57.80654	10.07742
0.13	125.8403	0.644097	202.0344	72.92252	10.4275
0.14	135.5241	0.555059	212.0252	82.60541	10.5274
0.15	145.2041	0.480712	220.5574	91.22416	10.4272
0.16	154.8848	0.425208	227.7425	104.4854	10.09628
0.17	164.5651	0.378652	233.8716	113.5283	9.431526
0.18	174.2454	0.338964	239.7572	124.9802	8.497104
0.19	183.9257	0.301800	242.2919	153.478	7.294203
0.2	193.606	0.272121	245.4223	174.3061	5.821842
0.21	203.2862	0.246220	247.2583	197.5425	4.077825
0.22	212.9665	0.224901	248.4459	223.2885	2.059227
0.23	222.647	0.205770	248.7293	251.6476	-0.23639
0.240000	232.3272	0.188973	249.3522	282.7251	-2.81377
0.25	242.0075	0.174183	247.3216	315.6282	-5.67690
0.26	251.6878	0.161024	245.597	352.4653	-8.83029
0.27	261.3681	0.149217	242.349	392.2407	-12.2791
0.28	271.0484	0.138940	240.1661	435.2779	[illegible]
0.29	280.7287	0.129432	237.2592	482.6701	[illegible]
0.3	290.409	0.120947	233.7781	533.3525	-24.4482
0.31	300.0893	0.11327	229.8752	589.2148	-29.1235
0.32	309.7696	0.106301	225.217	642.277	-34.1334
0.33	319.4499	0.099956	220.5720	702.7523	-39.4543
0.34	329.1302	0.094153	215.7245	767.0225	-45.1284
0.349999	338.8105	0.088859	210.4571	836.2948	-51.1210
0.359999	348.4907	0.083991	205.4441	907.6372	-57.4792
0.369999	358.171	0.079512	200.0591	984.0457	-64.1765
0.379999	367.8513	0.075392	194.6057	1064.737	-71.2254
0.389999	377.5316	0.071596	189.0868	1149.781	-78.6115
0.399999	387.2119	0.068022	183.5419	1239.262	-86.4426
0.409999	396.8922	0.064784	177.9989	1333.388	-94.8193
0.419999	406.5725	0.061707	172.4817	1432.22	-103.175
0.429999	416.2528	0.058871	167.0119	1536.682	-112.125
0.439999	425.9331	0.056227	161.608	1645.602	-121.475
0.449999	435.6134	0.053774	156.1281	1759.401	-131.203
0.459999	445.2937	[illegible]	[illegible]	1878.495	[illegible]
0.469999	454.974	[illegible]	147.2278	2002.77	[illegible]
0.479999	464.6543	0.047242	[illegible]	2132.727	[illegible]

ORIGINAL PAGE IS
OF POOR QUALITY

30,000 Feet

0.489999	474.3346	0.045226	136.0534	2258.054	-174.534
0.499999	484.0148	0.043541	131.2995	2409.114	-186.468
0.509999	493.6952	0.041850	126.6773	2555.904	-198.863
0.519999	503.3755	0.040256	122.1893	2708.570	-211.728
0.529999	513.0557	0.038751	117.8366	2867.246	-225.074
0.539999	522.7361	0.037329	113.6197	3032.033	-238.909
0.549999	532.4163	0.035984	109.5379	3203.045	-253.242
0.559999	542.0966	0.034710	105.5899	3380.402	-268.085
0.569999	551.7769	0.033503	101.7739	3564.216	-283.445
0.579999	561.4571	0.032358	98.08728	3754.604	-299.332
0.589999	571.1375	0.031270	94.52761	3951.68	-315.757
0.599999	580.8178	0.030236	91.0916	4155.558	-332.728
0.609999	590.4981	0.029253	87.77604	4366.356	-350.256
0.619999	600.1784	0.028317	84.57747	4584.187	-368.350
0.629999	609.8586	0.027425	81.49225	4809.166	-387.020
0.639999	619.539	0.026575	78.51679	5041.408	-406.275
0.649999	629.2192	0.025763	75.64733	5281.029	-426.126
0.659999	638.8995	0.024989	72.88028	5528.142	-446.582
0.669999	648.5799	0.024245	70.21191	5782.865	-467.653
0.679999	658.2601	0.023540	67.63863	6045.315	-489.348
0.689999	667.9404	0.022862	65.1568	6315.601	-511.678
0.699999	677.6207	0.022214	62.76202	6593.841	-534.651
0.709999	687.301	0.021592	60.45223	6880.151	-558.275
0.719999	696.9813	0.020997	58.22607	7174.647	-582.569
0.729999	706.6615	0.020426	56.07639	7477.437	-607.532
0.739999	716.3418	0.019878	54.0017	7788.646	-633.179
0.749999	726.0221	0.019351	51.99901	8108.384	-659.517
0.759999	735.7024	0.018845	50.06548	8436.768	-686.558
0.769999	745.3827	0.018359	48.19827	8773.913	-714.311
0.779999	755.063	0.017891	46.39474	9119.93	-742.784
0.789999	764.7433	0.017441	44.65231	9474.941	-771.989
0.799999	774.4236	0.017008	42.96851	9839.055	-801.934
0.809999	784.1039	0.016590	41.34095	10212.39	-832.63
0.819999	793.7842	0.016188	39.76742	10595.06	-864.085
0.829999	803.4645	0.015800	38.24572	10987.19	-896.31
0.839999	813.1447	0.015428	36.77277	11388.88	-929.314
0.849999	822.825	0.015068	35.34954	11800.24	-963.106
0.859999	832.5053	0.014717	33.97118	12221.41	-997.697
0.869999	842.1856	0.014381	32.63685	12652.5	-1033.09
0.879999	851.8659	0.014055	31.34483	13093.6	-1069.31
0.889999	861.5462	0.013742	30.09339	13544.85	-1106.35
0.899999	871.2265	0.013438	28.88101	14006.36	-1144.22
0.909999	880.9068	0.013144	27.70512	14478.24	-1182.95
0.919999	890.5871	0.012860	26.56731	14960.61	-1222.54
0.929999	900.2672	0.012585	25.46315	15453.56	-1262.99
0.939999	909.9475	0.012318	24.39234	15957.27	-1304.30
0.949999	919.6279	0.012061	23.35359	16471.79	-1346.51
0.959999	929.3082	0.011811	22.3457	16997.25	-1389.81
0.969999	938.9885	0.011568	21.36748	17532.8	-1433.61
0.979999	948.6688	0.011334	20.41784	18081.22	-1478.53
0.989999	958.3491	0.011105	19.49559	18640.53	-1524.26
0.999999	968.0294	0.010885	18.6001	19210.95	-1571.12
1.009999	977.7095	0.010671	17.72	19792.89	-1618.34
1.019999	987.3899	0.010462	16.8[illegible]	20385.47	-[illegible]67.21
1.029999	997.0702	0.010259	16.[illegible]27	209[illegible]	-1717.12
1.039999	1006.751	0.010064	15.28077	21605.02	-1767.72

50,000 Feet

1.049999	1016.421	0.009873	14.48587	22238.22	-1819.29
1.059999	1026.111	0.009667	13.73124	22879.52	-1871.85
1.069999	1035.791	0.009507	12.99609	23533.03	-1925.41
1.079999	1045.472	0.009332	12.28072	24198.88	-1979.98
1.089999	1055.152	0.009161	11.5845	24877.17	-2035.56
1.099999	1064.832	0.008996	10.90669	25568.02	-2092.17

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Bank Ang 7

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0.01	9.6802	109.2525	7.826131	416.0869	-33.4277
0.02	19.3605	27.21315	21.08445	208.1875	-15.3164
0.03	29.0409	12.13916	37.52711	139.2078	-6.32365
0.04	38.7212	6.828285	55.71435	105.2462	-4.05480
0.05	48.4015	4.370104	74.72559	85.6144	-0.89136
0.06	58.0818	3.034794	93.88055	73.4933	1.668956
0.07	67.7621	2.229645	112.8567	66.02621	3.817303
0.08	77.44239	1.707072	130.659	61.64238	5.622512
0.089999	87.12257	1.348758	147.6007	60.22154	7.152274
0.099999	96.80299	1.092225	162.2925	60.81261	8.283572
0.11	106.4803	0.902914	177.6081	63.28865	9.330312
0.12	116.1535	0.758698	190.5097	67.65175	10.0403
0.13	125.8439	0.646457	201.995	74.20262	10.46142
0.14	135.5242	0.557411	212.103	82.42271	10.61595
0.15	145.2045	0.485567	220.8995	92.58097	10.50615
0.16	154.8848	0.425787	228.4662	104.6755	10.13385
0.17	164.5651	0.378036	234.8928	118.8379	9.500574
0.18	174.2454	0.337192	240.2708	135.1292	8.507175
0.19	183.9257	0.302638	244.6895	153.6382	7.453711
0.2	193.606	0.273131	248.2225	174.4582	6.039355
0.21	203.2863	0.247733	250.9769	197.6875	4.36242
0.22	212.9666	0.225728	252.9919	223.4268	2.420278
0.23	222.647	0.206526	254.3392	251.7798	0.205522
0.240000	232.3272	0.189574	255.0736	282.8518	-2.27400
0.25	242.0075	0.174604	255.2432	316.7499	-5.02510
0.26	251.6878	0.161516	254.8909	353.5822	-8.07915
0.27	261.3681	0.149862	254.0546	393.4584	-11.4115
0.28	271.0484	0.139352	252.7583	436.4833	-15.0397
0.29	280.7287	0.129907	251.0649	482.7831	-18.9590
0.3	290.409	0.121391	248.972	532.4539	-23.2066
0.31	300.0893	0.113686	246.5176	585.613	-27.7592
0.32	309.7696	0.106692	243.728	642.372	-32.6340
0.33	319.4499	0.100322	240.6291	702.8445	-37.8282
0.34	329.1302	0.094509	237.2459	757.142	-43.3732
0.249999	338.8105	0.089185	233.6033	835.3812	-49.2821
0.359999	348.4907	0.084299	229.7228	907.6717	-55.4985
0.369999	358.171	0.079804	225.6281	984.1292	-62.0921
0.379999	367.8512	0.075659	221.3642	1064.857	-69.0514
0.339999	377.5315	0.071829	216.9282	1149.999	-76.3837
0.399999	387.2119	0.068282	212.3534	1239.639	-84.0954
0.409999	396.8922	0.064992	207.6627	1333.902	-92.1970
0.419999	406.5725	0.061934	202.8782	1432.902	-100.693
0.429999	416.2528	0.059087	198.0211	1535.784	-109.552
0.439999	425.9331	0.056431	193.1117	1645.571	-118.902
0.449999	435.6134	0.053951	188.189	1759.468	-128.527
0.459999	445.2937	0.051631	183.211	1878.591	-138.768
0.469999	454.974	0.049457	178.254	2002.963	-149.372

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50,000 feet

0.479999	464.5542	0.047418	173.3133	2132.79	-150.408
0.489999	474.3346	0.045502	168.4028	2268.156	-171.891
0.499999	484.0148	0.043701	163.5348	2409.175	-183.834
0.509999	493.6952	0.042004	158.7203	2555.964	-195.244
0.519999	503.3755	0.040404	153.9692	2708.636	-209.132
0.529999	513.0557	0.038893	149.2899	2867.306	-222.504
0.539999	522.7361	0.037465	144.6897	3032.09	-235.370
0.549999	532.4163	0.036116	140.1747	3203.1	-250.725
0.559999	542.0966	0.034838	135.75	3380.456	-265.620
0.569999	551.7769	0.033626	131.4157	3564.263	-281.022
0.579999	561.4571	0.032477	127.1855	3754.655	-296.954
0.589999	571.1275	0.031385	123.054	3951.731	-313.425
0.599999	580.8178	0.030247	119.0229	4155.609	-330.446
0.609999	590.4981	0.029361	115.0944	4366.407	-348.024
0.619999	600.1784	0.028421	111.269	4584.236	-366.169
0.629999	609.8586	0.027525	107.5462	4809.215	-384.891
0.639999	619.539	0.026673	103.927	5041.456	-404.199
0.649999	629.2192	0.025858	100.409	5281.075	-424.102
0.659999	638.8995	0.025080	96.99149	5528.135	-444.612
0.669999	648.5798	0.024337	93.67302	5782.911	-465.736
0.679999	658.2601	0.023627	90.45191	6045.359	-487.484
0.689999	667.9404	0.022947	87.32522	6315.645	-509.867
0.699999	677.6207	0.022295	84.29401	6592.884	-532.892
0.709999	687.301	0.021672	81.35305	6880.194	-556.571
0.719999	696.9813	0.021074	78.50114	7174.688	-580.912
0.729999	706.6616	0.020501	75.73593	7477.48	-605.927
0.739999	716.3418	0.019951	73.05517	7788.585	-631.622
0.749999	726.0221	0.019422	70.45621	8108.425	-658.01
0.759999	735.7024	0.018914	67.93709	8436.609	-685.098
0.769999	745.3827	0.018426	65.49507	8773.951	-712.898
0.779999	755.063	0.017957	63.12783	9119.97	-741.417
0.789999	764.7433	0.017505	60.83311	9474.979	-770.667
0.799999	774.4235	0.017070	58.60657	9839.092	-800.657
0.809999	784.1038	0.016651	56.4519	10212.43	-831.395
0.819999	793.7842	0.016248	54.36087	10595.1	-862.892
0.829999	803.4645	0.015859	52.33371	10987.22	-895.155
0.839999	813.1447	0.015483	50.35709	11388.91	-928.204
0.849999	822.825	0.015121	48.46017	11800.29	-962.035
0.859999	832.5053	0.014771	46.61046	12221.45	-996.665
0.869999	842.1856	0.014434	44.816	12652.53	-1032.10
0.879999	851.8659	0.014108	43.07498	13093.64	-1068.35
0.889999	861.5462	0.013792	41.38546	13544.89	-1105.43
0.899999	871.2265	0.013487	39.74574	14006.4	-1143.34
0.909999	880.9068	0.013193	38.15403	14478.27	-1182.10
0.919999	890.5871	0.012907	36.60969	14960.54	-1221.72
0.929999	900.2673	0.012631	35.1061	15453.81	-1262.20
0.939999	909.9476	0.012364	33.65073	15957.3	-1303.55
0.949999	919.6279	0.012105	32.23505	16471.83	-1345.79
0.959999	929.3082	0.011854	30.85957	16997.3	-1388.91
0.969999	938.9885	0.011611	29.52315	17532.83	-1432.95
0.979999	948.6688	0.011375	28.22417	18081.55	-1477.89
0.989999	958.3491	0.011147	26.96145	18640.55	-1523.75
0.999999	968.0294	0.010925	25.73371	19210.98	-1570.55
1.009999	977.7096	[illegible]	24.5398	19792.58	-1618.29
1.019999	987.3899	[illegible]	23.37853	20385.5	-1666.98
1.029999	997.0702	0.010295	22.24884	20991.64	-1716.62

50,000 feet

1.039999	1006.751	0.010101	21.1496	21609.05	-1767.24
1.049999	1016.431	0.009909	20.07982	22238.25	-1818.84
1.059999	1026.111	0.009723	19.03851	22879.55	-1871.42
1.069999	1035.791	0.009542	18.02477	23533.06	-1925.00
1.079999	1045.472	0.009366	17.03756	24198.91	-1979.59
1.089999	1055.152	0.009195	16.07611	24877.2	-2035.2
1.099999	1064.832	0.009029	15.13937	25568.05	-2091.83

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Bank Ang 9

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0.01	9.6803	109.7899	7.607205	420.1898	-23.7751
0.02	19.3606	27.44748	21.01095	210.2389	-15.4907
0.03	29.0409	12.19888	37.42907	140.5755	-8.44384
0.04	38.7212	6.861871	55.56415	106.2719	-4.15107
0.05	48.4015	4.391598	74.51745	86.43499	-0.97559
0.06	58.0818	3.049721	93.61082	74.17712	1.590696
0.07	67.7621	2.240611	112.3239	66.61232	3.74207
0.08	77.44233	1.715468	130.2542	62.35584	5.559311
0.089999	87.12259	1.355432	147.1462	60.58743	7.077335
0.099999	96.80295	1.0979	162.787	61.22391	8.314234
0.11	106.4833	0.907354	177.0697	63.78165	9.275708
0.12	116.1636	0.762430	189.9512	68.20256	9.966595
0.13	125.8439	0.649644	201.426	74.21222	10.35
0.14	135.5242	0.560152	211.5741	82.71578	10.54669
0.15	145.2045	0.487955	220.4299	92.83449	10.44531
0.16	154.8848	0.428866	228.0519	104.9319	10.08221
0.17	164.5651	0.379895	234.6544	119.0792	9.4613
0.18	174.2454	0.338857	240.2133	135.3572	8.583809
0.19	183.9257	0.304127	244.9519	153.8541	7.45015
0.2	193.606	0.274474	248.6874	174.6634	6.059804
0.21	203.2863	0.248956	251.7697	197.8829	4.411322
0.22	212.9666	0.226839	254.1806	223.6133	2.502218
0.23	222.647	0.207542	255.9836	251.9582	0.329527
0.240000	232.3273	0.190607	257.2344	283.0228	-2.11110
0.25	242.0076	0.175662	257.9516	316.914	-4.82427
0.26	251.6879	0.162411	258.257	353.7401	-7.81389
0.27	261.3682	0.150603	258.127	392.6104	-11.0910
0.28	271.0484	0.140036	257.5934	435.635	-14.6656
0.29	280.7287	0.130546	256.6937	482.9246	-18.5198
0.3	290.409	0.121986	255.4524	532.5907	-22.6873
0.31	300.0893	0.114245	253.8912	585.7453	-27.1664
0.32	309.7696	0.107215	252.0299	642.5003	-31.9649
0.33	319.4499	0.100817	249.8668	702.9689	-37.0905
0.34	329.1302	0.094974	247.479	767.2528	-42.5510
0.349999	338.8105	0.089624	244.8229	835.4994	-48.3543
0.359999	348.4907	0.084714	241.9342	907.7858	-54.5084
0.369999	358.171	0.080197	238.8286	984.24	-61.0213
0.379999	367.8512	0.076031	235.5218	1064.975	-67.9012
0.389999	377.5316	0.072182	232.0288	1150.104	-75.1561
0.399999	387.2119	0.068618	228.3549	1239.742	-82.7940
0.409999	396.8922	0.065312	224.5458	1334.003	-90.8221
0.419999	406.5725	0.062239	220.5852	1433	-99.2514
0.429999	416.2528	0.059378	216.5121	1536.849	-108.087
0.439999	425.9331	0.056709	212.3103	1645.552	-117.328
0.449999	435.6134	0.054217	208.0222	1759.22	-127.012
0.459999	445.2937	0.051886	203.6666	1878.85	-137.115

50,000 feet

0.459999	454.974	0.049701	199.2251	2003.051	-147.655
0.479999	464.6543	0.047651	194.7427	2132.876	-158.660
0.489999	474.3346	0.045725	190.2228	2268.24	-170.112
0.499999	484.0148	0.043916	185.6785	2409.257	-182.028
0.509999	493.6952	0.042210	181.1218	2556.045	-194.417
0.519999	503.3755	0.040602	176.5643	2708.714	-207.288
0.529999	513.0557	0.039085	172.0169	2867.383	-220.650
0.539999	522.7361	0.037650	167.4895	3032.155	-234.510
0.549999	532.4152	0.036294	162.9912	3203.175	-248.877
0.559999	542.0966	0.035009	158.5305	3380.529	-263.761
0.569999	551.7769	0.033791	154.1149	3564.341	-279.170
0.579999	561.4271	0.032636	149.7512	3754.727	-295.113
0.589999	571.1375	0.031539	145.4456	3951.801	-311.598
0.599999	580.8178	0.030497	141.2032	4155.677	-328.635
0.609999	590.4981	0.029505	137.0285	4366.474	-346.234
0.619999	600.1784	0.028561	132.9256	4584.302	-364.401
0.629999	609.8588	0.027661	128.8978	4809.28	-383.148
0.639999	619.539	0.026804	124.9475	5041.25	-402.424
0.649999	629.2195	0.025985	121.0772	5281.139	-422.416
0.659999	638.8995	0.025204	117.2884	5528.251	-442.955
0.669999	648.5795	0.024457	113.5823	5782.973	-464.111
0.679999	658.2601	0.023743	109.9595	6045.419	-485.899
0.689999	667.9404	0.023060	106.431	6315.704	-508.306
0.699999	677.6207	0.022408	102.9664	6593.944	-531.369
0.709999	687.301	0.021779	99.59552	6880.253	-555.083
0.719999	696.9812	0.021178	96.30807	7174.745	-579.460
0.729999	706.6615	0.020602	93.10325	7477.535	-604.509
0.739999	716.3418	0.020049	89.98024	7788.743	-630.241
0.749999	726.0221	0.019518	86.93786	8108.479	-656.655
0.759999	735.7024	0.019007	83.975	8436.862	-683.790
0.769999	745.3827	0.018517	81.09026	8774.006	-711.625
0.779999	755.063	0.018045	78.28225	9120.022	-740.181
0.789999	764.7433	0.017591	75.54946	9475.031	-769.467
0.799999	774.4236	0.017154	72.89021	9839.144	-799.492
0.809999	784.1039	0.016733	70.30295	10212.48	-830.265
0.819999	793.7842	0.016328	67.78592	10595.15	-861.793
0.829999	803.4645	0.015937	65.33752	10987.27	-894.099
0.839999	813.1447	0.015559	62.95595	11388.96	-927.177
0.849999	822.825	0.015195	60.63949	11800.33	-961.043
0.859999	832.5053	0.014844	58.3864	12221.5	-995.705
0.869999	842.1855	0.014505	56.19494	12652.58	-1031.17
0.879999	851.8659	0.014177	54.06351	13093.66	-1067.45
0.889999	861.5462	0.013860	51.99024	13544.92	-1104.55
0.899999	871.2265	0.013554	49.97362	14006.44	-1142.51
0.909999	880.9068	0.013258	48.01197	14478.32	-1181.20
0.919999	890.5871	0.012971	46.10262	14960.66	-1220.94
0.929999	900.2673	0.012693	44.24703	15453.55	-1261.45
0.939999	909.9476	0.012425	42.44063	15957.34	-1302.83
0.949999	919.6279	0.012165	40.68297	16471.97	-1345.10
0.959999	929.3082	0.011912	38.97247	16997.34	-1388.25
0.969999	938.9885	0.011668	37.30775	17533.88	-1432.31
0.979999	948.6688	0.011431	35.68742	18081.59	-1477.29
0.989999	958.3491	0.011201	34.11009	18640.6	-1523.17
0.999999	968.0294	0.010978	32.57443	19211.02	-1570
1.009999	977.7096	0.010762	31.07920	19793.46	-1617.78
1.019999	987.3899	0.010552	29.62213	20386.34	-1666.47

30,000 feet

1.029999	997.0702	0.010348	28.20501	20991.88	-1716.14
1.039999	1006.751	0.010150	26.82363	21609.09	-1766.78
1.049999	1016.431	0.009958	25.4778	22238.29	-1818.40
1.059999	1026.111	0.009771	24.16655	22879.59	-1871.00
1.069999	1035.791	0.009589	22.88877	23533.1	-1924.51
1.079999	1045.472	0.009412	21.6434	24198.94	-1979.22
1.089999	1055.152	0.009240	20.42948	24877.24	-2034.84
1.099999	1064.832	0.009073	19.24593	25568.05	-2091.49

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Bank Ang 11

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0.01	9.6803	110.4678	7.583524	425.3946	-34.2031
0.02	19.3606	27.61696	20.94403	212.6414	-15.7092
0.03	29.0409	12.27421	37.30626	142.2104	-8.59591
0.04	38.7212	6.90424	55.37595	107.5731	-4.27259
0.05	48.4015	4.418715	74.25625	87.47594	-1.0822
0.06	58.0818	3.068522	93.27094	75.04458	1.49808
0.07	67.7621	2.254446	111.9018	67.35058	3.648550
0.08	77.44239	1.726051	129.7589	63.00584	5.454564
0.089999	87.12259	1.363801	146.5612	61.26573	6.982524
0.099999	96.80299	1.104579	162.1222	61.74438	8.217198
0.11	106.4832	0.912857	176.2244	64.22421	9.178774
0.12	116.1636	0.767134	189.1557	68.52728	9.865981
0.13	125.8439	0.653654	200.5955	74.91862	10.28625
0.14	135.5242	0.563611	210.7008	83.08754	10.44577
0.15	145.2045	0.490968	219.5451	93.18148	10.34447
0.16	154.8848	0.431515	227.2178	105.2572	9.932999
0.17	164.5651	0.382241	233.8149	119.3854	9.367512
0.18	174.2454	0.340950	239.4355	135.6452	8.495454
0.19	183.9257	0.306005	244.1747	154.128	7.371466
0.2	193.606	0.276169	248.1218	174.9236	5.992202
0.21	203.2863	0.250493	251.3585	198.1307	4.357373
0.22	212.9666	0.228239	253.9584	223.8499	2.464751
0.23	222.647	0.208822	255.9851	252.1845	0.311212
0.240000	232.3273	0.191784	257.4966	283.2256	-2.10722
0.25	242.0076	0.176746	258.5488	317.1202	-4.79287
0.26	251.6878	0.163412	259.1682	353.9404	-7.75829
0.27	261.3681	0.151537	259.4047	393.8021	-11.0722
0.28	271.0484	0.140908	259.2855	436.8209	-14.6335
0.29	280.7287	0.131353	258.8378	483.1041	-18.3590
0.3	290.409	0.122742	258.0848	532.7542	-22.4850
0.31	300.0893	0.114950	257.0469	585.9131	-26.9218
0.32	309.7696	0.107878	255.7414	642.6528	-31.6744
0.33	319.4499	0.101439	254.1843	703.1285	-36.7516
0.34	329.1302	0.095559	252.3894	767.4168	-42.1615
0.349999	338.8105	0.090177	250.3696	835.6471	-47.9124
0.359999	348.4907	0.085237	248.1268	907.9302	-54.0125
0.369999	358.171	0.080692	245.7023	984.3807	-60.4702
0.379999	367.8513	0.076501	243.0759	1065.112	-67.2940
0.389999	377.5316	0.072623	240.2712	1150.238	-74.4923
0.399999	387.2119	0.069042	237.2956	1239.872	-82.0735
0.409999	396.8922	0.065715	234.1604	1334.129	-90.0464
0.419999	406.5725	0.062623	230.8751	1433.124	-98.4192
0.429999	416.2528	0.059744	227.4502	1536.97	-107.200
0.439999	425.9331	0.057059	223.9022	1645.782	-116.399
0.449999	435.6134	0.054552	220.2341	1759.675	-126.022

ORIGINAL PAGE IS
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50,000 feet

0.459999	445.2937	0.052206	215.4593	1878.754	-135.080
0.469999	454.974	0.050008	212.5888	2003.162	-146.581
0.479999	464.6543	0.047946	208.6332	2132.984	-157.532
0.489999	474.3346	0.046009	204.6034	2268.346	-168.943
0.499999	484.0148	0.044187	200.51	2409.361	-180.822
0.509999	493.6952	0.042471	196.3635	2556.146	-193.178
0.519999	503.3755	0.040853	192.1741	2708.815	-206.019
0.529999	513.0557	0.039325	187.9518	2867.482	-219.353
0.539999	522.7361	0.037883	183.7061	3032.252	-233.190
0.549999	532.4163	0.036518	179.4465	3203.27	-247.532
0.559999	542.0966	0.035225	175.1815	3380.622	-262.405
0.569999	551.7769	0.034000	170.9158	3564.423	-277.802
0.579999	561.4571	0.032838	166.6591	3754.817	-293.735
0.589999	571.1375	0.031734	162.4368	3951.889	-310.214
0.599999	580.8178	0.030685	158.2298	4155.764	-327.249
0.609999	590.4981	0.029687	154.0545	4366.558	-344.847
0.619999	600.1784	0.028737	149.9166	4584.385	-363.017
0.629999	609.8586	0.027832	145.8214	4809.362	-381.770
0.639999	619.539	0.026969	141.7728	5041.602	-401.113
0.649999	629.2192	0.026146	137.7778	5281.219	-421.055
0.659999	638.8995	0.025359	133.8374	5528.33	-441.607
0.669999	648.5798	0.024602	129.9557	5783.052	-462.777
0.679999	658.2601	0.023880	126.1355	6045.495	-484.575
0.689999	667.9404	0.023202	122.3798	6315.78	-507.008
0.699999	677.6207	0.022544	118.6901	6594.017	-530.088
0.709999	687.301	0.021913	115.068	6880.325	-553.822
0.719999	696.9812	0.021309	111.5151	7174.817	-578.221
0.729999	706.6615	0.020729	108.0323	7477.608	-603.293
0.739999	716.3418	0.020172	104.6203	7788.812	-629.046
0.749999	726.0221	0.019638	101.2794	8108.549	-655.495
0.759999	735.7024	0.019125	98.01006	8436.921	-682.645
0.769999	745.3827	0.018631	94.81209	8774.072	-710.508
0.779999	755.063	0.018157	91.68528	9120.089	-739.089
0.789999	764.7433	0.017700	88.62922	9475.097	-768.401
0.799999	774.4235	0.017260	85.64344	9839.21	-798.452
0.809999	784.1039	0.016837	82.7271	10212.54	-829.254
0.819999	793.7842	0.016429	79.87952	10595.21	-860.818
0.829999	803.4645	0.016035	77.09972	10987.32	-893.141
0.839999	813.1447	0.015655	74.38555	11389.02	-926.245
0.849999	822.825	0.015289	71.73935	11800.39	-960.129
0.859999	832.5053	0.014936	69.15654	12221.56	-994.859
0.869999	842.1856	0.014594	66.63707	12652.64	-1030.32
0.879999	851.8659	0.014264	64.17973	13093.74	-1066.63
0.889999	861.5462	0.013946	61.78318	13544.99	-1103.77
0.899999	871.2265	0.013638	59.44615	14005.5	-1141.74
0.909999	880.9068	0.013339	57.1673	14478.37	-1180.55
0.919999	890.5871	0.013051	54.94523	14960.74	-1220.22
0.929999	900.2673	0.012772	52.77884	15453.7	-1260.75
0.939999	909.9476	0.012502	50.66652	15957.4	-1302.15
0.949999	919.6279	0.012240	48.60705	16471.92	-1344.45
0.959999	929.3082	0.011986	46.59913	16997.39	-1387.63
0.969999	938.9885	0.011740	44.64129	17533.93	-1431.72
0.979999	948.6688	0.011502	42.7325	18081.54	-1476.71
0.989999	958.349	0.011271	40.87183	18640.85	-1522.62
0.999999	968.0294	0.011046	39.05827	19211.07	-1569.47
1.009999	977.7096	0.010829	37.28522	19793.01	-1617.25

50,000 feet

1.019999	987.3895	0.010617	35.58059	20385.59	-1565.99
1.029999	997.0702	0.010412	33.87735	20991.93	-1715.68
1.039999	1006.751	0.010213	32.23559	21609.14	-1766.34
1.049999	1016.431	0.010019	30.6342	22238.34	-1817.58
1.059999	1026.111	0.009831	29.07202	22879.64	-1870.60
1.069999	1035.791	0.009648	27.54801	23533.15	-1924.23
1.079999	1045.472	0.009470	26.06102	24198.99	-1978.86
1.089999	1055.152	0.009297	24.61008	24877.29	-2034.50
1.099999	1064.832	0.009129	23.19416	25568.13	-2091.17

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Bank Ang 13

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0.01	9.6803	111.2906	7.555075	431.7548	-24.7261
0.02	19.3606	27.82265	20.85362	216.0214	-15.9761
0.03	29.0409	12.36563	37.15692	144.4304	-8.78153
0.04	38.7212	6.955563	55.14578	109.1531	-4.42167
0.05	48.4015	4.451625	73.94204	88.74792	-1.21209
0.06	58.0818	3.091406	92.65151	75.10438	1.271777
0.07	67.7621	2.271227	111.3928	68.28444	3.520582
0.08	77.44225	1.732915	129.147	63.50084	5.349415
0.089999	87.12229	1.372559	145.2488	61.5724	6.866191
0.099999	96.80[illegible]	[illegible]	[illegible]	62.32029	8.099451
0.11	106.4832	0.919757	170.425	64.31256	9.052001
0.12	116.1636	0.772881	186.1591	69.1574	9.740985
0.13	125.8439	0.652524	199.5218	75.40785	10.1603
0.14	135.5242	0.557509	209.5524	83.54183	10.31529
0.15	145.2045	0.494825	218.2562	92.60548	10.21244
0.16	154.8848	0.434729	225.9945	105.6547	9.85134
0.17	164.5651	0.385088	232.5755	119.7595	9.235421
0.18	174.2454	0.343499	233.1989	135.9997	8.258305
0.19	183.9257	0.308284	242.9613	154.4627	7.244735
0.2	193.606	0.278225	246.9532	175.2416	5.870504
0.21	203.2863	0.252359	250.2572	198.4336	4.242425
0.22	212.9666	0.229929	252.9477	224.139	2.358357
0.23	222.647	0.210273	255.0903	252.461	0.215229
0.240000	232.3273	0.193213	256.7425	283.5047	-2.19132
0.25	242.0076	0.178064	257.9540	317.3756	-4.86447
0.26	251.6879	0.164621	258.7567	354.1849	-7.81103
0.27	261.3681	0.152582	259.2225	394.0387	-11.0264
0.28	271.0484	0.141952	259.3475	437.048	-14.5470
0.29	280.7287	0.132331	259.1704	483.3234	-18.3497
0.3	290.409	0.123555	258.7146	532.9752	-22.4518
0.31	300.0893	0.115807	257.9996	586.1183	-26.8505
0.32	309.7696	0.108562	257.0425	642.8615	-31.5849
0.33	319.4499	0.102195	255.9583	703.3192	-36.6303
0.34	329.1302	0.096272	254.4595	767.6028	-42.0074
0.349999	338.8105	0.090849	252.8578	835.8287	-47.7235
0.359999	348.4907	0.085972	251.0533	908.1059	-53.7272
0.369999	358.171	0.081293	249.0853	984.5525	-60.2073
0.379999	367.8513	0.077071	246.9324	1065.279	-66.9921
0.389999	377.5316	0.073183	244.6127	1150.401	-74.1502
0.399999	387.2119	0.069555	242.134	1240.031	-81.6902
0.409999	396.8922	0.066202	239.5037	1334.282	-89.6217
0.419999	406.5725	[illegible]	236.7295	1433.275	-97.9525
0.429999	416.2528	0.060183	233.8187	1537.112	-106.631
0.439999	425.9331	0.057464	230.7738	1645.927	-115.647

ORIGINAL PAGE IS
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0.449999	435.6124	0.054958	227.6172	1759.817	-125.43
0.459999	445.2937	0.052594	224.3417	1878.902	-135.446
0.469999	454.974	0.050380	220.96	2003.297	-145.907
0.479999	464.6543	0.048303	217.4789	2133.117	-156.819
0.489999	474.3346	0.046351	213.9096	2268.475	-168.192
0.499999	484.0148	0.044516	210.2569	2409.488	-180.035
0.509999	493.6952	0.042787	206.53	2556.271	-192.255
0.519999	503.3755	0.041157	202.737	2708.937	-205.184
0.529999	513.0557	0.039619	198.8851	2867.602	-218.458
0.539999	522.7361	0.038165	194.9854	3032.28	-232.276
0.549999	532.4163	0.036790	191.0423	3203.325	-246.592
0.559999	542.0966	0.035488	187.0561	3380.736	-261.442
0.569999	551.7769	0.034253	183.063	3564.544	-276.817
0.579999	561.4571	0.033082	179.0409	3754.926	-292.731
0.589999	571.1375	0.031970	175.0071	3951.995	-309.194
0.599999	580.8178	0.030914	170.9665	4155.871	-326.215
0.609999	590.4981	0.029908	166.9215	4366.552	-343.801
0.619999	600.1784	0.028951	162.9023	4584.487	-361.960
0.629999	609.8587	0.028039	158.8881	4809.452	-380.713
0.639999	619.539	0.027170	154.8931	5041.701	-400.047
0.649999	629.2192	0.026341	150.9229	5281.217	-419.988
0.659999	638.8995	0.025549	146.9934	5528.489	-440.527
0.669999	648.5798	0.024791	143.0761	5782.145	-461.7[illegible]
0.679999	658.2601	0.024068	139.2081	6045.589	-483.512
0.689999	667.9404	0.023375	135.3821	6315.672	-505.951
0.699999	677.6207	0.022712	131.6015	6594.109	-529.033
0.709999	687.301	0.022077	127.8693	6880.415	-552.781
0.719999	696.9812	0.021468	124.1862	7174.905	-577.191
0.729999	706.6615	0.020883	120.5605	7477.594	-602.273
0.739999	716.3418	0.020322	116.9892	7788.898	-628.043
0.749999	726.0221	0.019783	113.4731	8108.625	-654.505
0.759999	735.7024	0.019267	110.0167	8437.014	-681.670
0.769999	745.3827	0.018770	106.6202	8774.154	-709.546
0.779999	755.063	0.018292	103.2844	9120.17	-738.146
0.789999	764.7432	0.017832	[illegible]	9475.178	-767.475
0.799999	774.4235	0.017389	96.79772	9839.268	-797.547
0.809999	784.1039	0.016962	93.64765	10212.88	-828.368
0.819999	793.7841	0.016551	90.56047	10596.29	-859.949
0.829999	803.4644	0.016154	87.53551	10987.41	-892.293
0.839999	813.1447	0.015772	84.57279	11389.1	-925.419
0.849999	822.825	0.015403	81.57209	11800.47	-959.332
0.859999	832.5053	0.015047	78.83292	12221.63	-994.042
0.869999	842.1855	0.014703	75.05478	12652.71	-1029.55
0.879999	851.8659	0.014371	73.33705	13092.82	-1065.88
0.889999	861.5462	0.014050	70.67905	13545.05	-1103.05
0.899999	871.2265	0.013739	68.07956	14006.57	-1141.04
0.909999	880.9068	0.013439	65.53397	14478.44	-1179.88
0.919999	890.5871	0.013148	63.05517	14960.81	-1219.57
0.929999	900.2673	0.012867	60.62751	15452.78	-1260.12
0.939999	909.9476	0.012595	58.25532	15957.47	-1301.55
0.949999	919.6279	0.012331	55.93733	16471.59	-1342.86
0.959999	929.3082	0.012075	53.67255	16997.46	-1387.08
0.969999	938.9885	0.011827	51.46592	17533.99	-1431.18
0.979999	948.6688	0.011587	49.2984	18081.71	-1476.12
0.989999	958.349	0.011355	47.18896	18640.73	-1522.17
0.999999	968.0294	0.011123	45.12445	19211.13	-1568.78

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50,000 feet

1.009999	977.7095	0.010909	43.10973	19793.07	-1616.78
1.019999	987.3899	0.010696	41.14194	20288.66	-1665.53
1.029999	997.0702	0.010490	39.21975	20991.95	-1715.25
1.039999	1006.751	0.010289	37.34222	21609.2	-1765.93
1.049999	1016.431	0.010094	35.50825	22238.4	-1817.59
1.059999	1026.111	0.009904	33.71677	22879.7	-1870.23
1.069999	1035.791	0.009720	31.96561	23533.2	-1923.87
1.079999	1045.472	0.009541	30.25727	24199.05	-1978.52
1.089999	1055.152	0.009367	28.58781	24877.35	-2034.18
1.099999	1064.832	0.009197	26.9222	25568.2	-2090.87

Optimal Climb Conditions
Chosen from Climb Rate Analysis
Climb Rate maximized

Altitude 10^-3 ft	Mach #	Velocity ft/s	Bank Ang degrees	C lift	Climb ft/s	Radius ft
5	0.06	65.826	9	0.421847	11.6018	850.3101
25	0.09	91.44	5	0.416338	11.88128	2970.404
50	0.14	135.52	5	0.555369	10.6274	6524.540
75	0.22	212.97	7	0.744382	7.575022	11481.22
100	0.38	381.216	9	0.802255	0.108634	28518.36

Number of turns
(change in altitude per turn)

dv/dh(x10^3) 1.2807
dr/dh(x10^3) 106.0047
((dh/dt)/dh)*10^3)0.013974

Altitude 10^-3 ft	Velocity ft/s	Radius ft	Omega	Time of Period	Climb ft/s	change in altitude
5	65.826	850.3101	0.077414	88.25075	11.6018	1.023867
6.023867	67.13726	958.8449	0.070018	97.57155	11.61610	1.133401
7.157269	68.58881	1078.990	0.063567	107.4738	11.63194	1.250130
8.407399	70.18985	1211.510	0.057935	117.9210	11.64941	1.373711
9.781111	71.94916	1357.130	0.053015	128.8648	11.66861	1.503673
11.28478	73.87492	1516.526	0.048713	140.2463	11.68962	1.639427
12.92421	75.97453	1690.313	0.044946	151.9979	11.71253	1.780281
14.70449	78.25454	1879.032	0.041646	164.0450	11.73741	1.925464
16.62995	80.72048	2083.140	0.038749	176.3085	11.76431	2.074149
18.70410	83.37684	2303.010	0.036203	188.7073	11.79330	2.225482
20.92958	86.22702	2538.921	0.033962	201.1613	11.82440	2.378611
23.30820	89.27331	2791.065	0.031985	213.5929	11.85763	2.532708

dv/dh(x10^3) 1.7632
dr/dh(x10^3) 142.1654
((dh/dt)/dh)*10^3)-0.05015

Altitude 10^-3 ft	Velocity ft/s	Radius ft	Omega rad vel	Time of Period	Climb ft/s	change in altitude
25	91.44	2970.404	0.030783	221.9309	11.88128	2.636823
27.63682	96.08924	3345.269	0.028723	237.8454	11.74902	2.794453
30.43127	101.0164	3742.544	0.026991	253.1124	11.60887	2.938350
33.36962	106.1973	4160.276	0.025526	267.6375	11.46149	3.067528
36.43715	111.6059	4596.372	0.024281	281.3625	11.30764	3.181548
39.61870	117.2156	5048.679	0.023217	294.2595	11.14807	3.280427
42.89913	122.9997	5515.042	0.022302	306.3255	10.98354	3.364540
46.26367	128.9321	5993.363	0.021512	317.5763	10.81479	3.434523
49.69819	134.9878	6481.634	0.020826	328.0411	10.64253	3.491190
53.18938	141.1435	6977.961	0.020227	337.7583	10.46743	3.535463

dv/dh(x10^3) 3.098
dr/dh(x10^3) 198.2671
((dh/dt)/dh)*10^3)-0.12209

Altitude 10^-3 ft	Velocity ft/s	Radius ft	Omega rad vel	Time of Period	Climb ft/s	change in altitude
50	135.52	6524.540	0.020770	328.9160	10.6274	3.495522
53.49552	146.3491	7217.588	0.020276	336.9306	10.20061	3.436899
56.93242	156.9966	7899.012	0.019875	343.7327	9.780985	3.362045
60.29446	167.4122	8565.595	0.019544	349.5496	9.370495	3.275453
63.56991	177.5596	9215.010	0.019268	354.5603	8.970578	3.180611
66.75053	187.4131	9845.621	0.019035	358.9067	8.582241	3.080224
69.83075	196.9556	10456.32	0.018836	362.7013	8.206161	2.976386

72.80714	206.1765	11046.44	0.018664	366.0344	7.842759	2.870720

75.67786	215.0700	11615.61	0.018515	368.9784	7.492258	2.764481

dv/dh(x10^3)	6.72984
dr/dh(x10^3)	681.4859
((dh/dt)/dh)*10^3)	-0.29865

Altitude 10^-3 ft	Velocity ft/s	Radius ft	Omega rad vel	Time of Period	Climb ft/s	change in altitude
75	212.97	11481.22	0.018549	368.3054	7.575022	2.789921
77.78992	231.7457	13382.51	0.017317	394.5158	6.741796	2.659745
80.44966	249.6453	15195.09	0.016429	415.8323	5.947448	2.473141
82.92280	266.2892	16880.50	0.015774	433.0821	5.208831	2.255851
85.17866	281.4707	18417.83	0.015282	447.0373	4.535108	2.027362
87.20602	295.1145	19799.45	0.014905	458.3540	3.929625	1.801159
89.00718	307.2361	21026.91	0.014611	467.5648	3.391699	1.585839
90.59302	317.9085	22107.64	0.014380	475.0931	2.918079	1.386359
91.97938	327.2385	23052.43	0.014195	481.2722	2.504035	1.205122
93.18450	335.3488	23873.70	0.014046	486.3641	2.144119	1.042822
94.22732	342.3668	24584.37	0.013926	490.5756	1.832674	0.899065
95.12639	348.4174	25197.07	0.013827	494.0703	1.564163	0.772806
95.89920	353.6182	25723.73	0.013746	496.9786	1.333360	0.662651
96.56185	358.0778	26175.31	0.013679	499.4052	1.135455	0.567052
97.12890	361.8939	26561.75	0.013624	501.4341	0.966102	0.484436
97.61334	365.1541	26891.89	0.013578	503.1339	0.821422	0.413285
98.02662	367.9355	27173.54	0.013540	504.5602	0.697992	0.352179
98.37880	370.3056	27413.54	0.013508	505.7587	0.592812	0.299820
98.67862	372.3233	27617.87	0.013481	506.7671	0.503269	0.255040
98.93366	374.0397	27791.67	0.013458	507.6162	0.427100	0.216803
99.15047	375.4987	27939.42	0.013439	508.3319	0.362350	0.184194
99.33466	376.7383	28064.95	0.013423	508.9357	0.307340	0.156416
99.49108	377.7910	28171.54	0.013410	509.4452	0.260625	0.132774
99.62385	378.6846	28262.03	0.013399	509.8756	0.220971	0.112668
99.73652	379.4428	28338.81	0.013389	510.2391	0.187322	0.095579
99.83210	380.0860	28403.94	0.013381	510.5464	0.158777	0.081063
99.91316	380.6316	28459.19	0.013374	510.8062	0.134567	0.068737
99.98190	381.0942	28506.03	0.013368	511.0259	0.114038	0.058276
100.0401	381.4864	28545.75	0.013364	511.2118	0.096633	0.049400
100	381.216	28518.36	0.013367	511.0837	0.108634	0.055521

9.5 V-n Diagram

```
100 REM
110 REM                 Effect of Gust
120 REM
130 REM
140 REM
150 REM @@@@@@@@@@@@@@@@@ Contants @@@@@@@@@@@@@@@@
160 REM
170            PI = 3.141592654#           :REM  Numerical value for pie
180            A0 = (2*PI)                :REM  Value for 2 * pie
190            C1 = 1.5                   :REM  Max C lift
200            C2 = -1                    :REM  Min C lift
210            I1 = .1                    :REM  Increment for Mach #
220            I2 = 10                    :REM  Increment for gust
230 REM
240 REM @@@@@@@@@@@@@@@@@ Input Variables @@@@@@@@@
250 REM
260            INPUT "Gross Weight                       ";W
270            INPUT "Wing Area                          ";S
280            INPUT "Aspect Ratio                       ";AR
290            INPUT "Altitude of flight                 ";FA
300            INPUT "Speed of sound @ altitude          ";SS
310            INPUT "Density @ Altitude                 ";RO
320 REM
330 REM @@@@@@@@@@@@@@@@ Calculations @@@@@@@@@@@@@
340 REM
350            OPEN "c:gustsl.out" FOR OUTPUT AS 1 LEN=2000
360 REM
370            FOR WG = -40 TO 40 STEP I2
380 REM
390                    PRINT "Wind gust velocity          ";WG
400 REM
410                    FOR M = .1 TO 1 STEP I1
420 REM
430                            V = M*SS
440 REM
450                            A = (A0/(1 + (A0/(PI*AR))))
460 REM
470                            N = 1 + ((RO*A*WG*V)/(2*(W/S)))
480                            N1 = (C1*.5*RO*(V^2)/(W/S))
490                            N2 = (C2*.5*RO*(V^2)/(W/S))
500 REM
510                            PRINT M;N1;N2;N
520                            PRINT #1, M;N1;N2;N
530 REM
540                    NEXT M
550 REM
560            NEXT WG
570 REM
580            CLOSE 1
590 REM
600            END
```

EFFECT OF WIND GUST ALTITUDE SEA LEVEL

MACH #	VERTICAL MAGNITUDE OF GUST (FT/S) 40	20	10	-10	-20	-4
0	1	1	1	1	1	
0.1	16.91369	8.956844	4.978422	-2.97842	-6.95684	-14.913
0.2	32.82738	16.91369	8.956844	-6.95684	-14.9136	-30.827
0.3	48.74107	24.87054	12.93527	-10.9352	-22.8705	-46.741
0.4	64.65476	32.82738	16.91369	-14.9136	-30.8273	-62.654
0.5	80.56845	40.78422	20.89211	-18.8921	-38.7842	-78.568
0.6	96.48214	48.74107	24.87054	-22.8705	-46.7410	-94.482
0.700000	112.3958	56.69792	28.84896	-26.8489	-54.6979	-110.39
0.800000	128.3095	64.65476	32.82738	-30.8273	-62.6547	-126.30
0.900000	144.2232	72.61161	36.80581	-34.8058	-70.6116	-142.[illegible]
1	160.1369	80.56845	40.78423	-38.7842	-78.5684	-158.13

LIMIT LOAD FACTOR CONDITIONS ALTITUDE SEA LEVEL
DUE TO MAX AND MIN CL'S

MACH #	CL = 1.5	CL = -1
0	0	0
0.1	12.0171	-8.01140
0.2	48.06841	-32.0456
0.3	108.1539	-72.1026
0.4	192.2736	-128.182
0.5	300.4276	-200.285
0.6	432.6157	-288.410
0.700000	588.8381	-392.558
0.800000	769.0948	-512.729
0.900000	973.3856	-648.923
1	1201.711	-801.140

EFFECT OF WIND GUST

ALTITUDE 25,000 (FEET)

VERTICAL MAGNITUDE OF GUST (FT/S)

MACH #	40	20	10	-10	-20	-4
0	1	1	1	1	1	
0.1	7.497005	4.248503	2.624251	-0.62425	-2.24850	-5.497
0.2	13.99401	7.497005	4.248503	-2.24850	-5.49700	-11.99
0.3	20.49102	10.74551	5.872754	-3.87275	-8.74550	-18.49
0.4	26.98802	13.99401	7.497005	-5.49700	-11.9940	-24.98
0.5	33.48503	17.24251	9.121257	-7.12125	-15.2425	-31.48
0.6	39.98203	20.49102	10.74551	-8.74550	-18.4910	-37.98
0.700000	46.47904	23.73952	12.36976	-10.3697	-21.7395	-44.47
0.800000	52.97605	26.98802	13.99401	-11.9940	-24.9880	-50.97
0.900000	59.47306	30.23653	15.61825	-13.6182	-28.2355	-57.47
1	65.97006	33.48503	17.24252	-15.2425	-31.4850	-63.9

LIMIT LOAD FACTOR CONDITIONS
DUE TO MAX AND MIN CL'S

ALTITUDE 25,000 (FEET)

MACH #	CL = 1.5	CL = -1
0	0	0
0.1	4.464943	-2.97662
0.2	17.85977	-11.9065
0.3	40.1845	-26.7896
0.4	71.43909	-47.6260
0.5	111.6236	-74.4157
0.6	160.738	-107.158
0.700000	218.7823	-145.854
0.800000	285.7564	-190.504
0.900000	361.6605	-241.107
1	446.4945	-297.653

EFFECT OF WIND GUST ALTITUDE 50,000 (FEET)

VERTICAL MAGNITUDE OF GUST (FT/S)

MACH #	40	20	10	-10	-20	-4
0	1	1	1	1	1	
0.1	3.11257	2.056285	1.528143	0.471857	-0.05628	-1.1125
0.2	5.225139	3.11257	2.056285	-0.05628	-1.11257	-3.225
0.3	7.337709	4.168855	2.584427	-0.58442	-2.16885	-5.3377
0.4	9.450278	5.225139	3.11257	-1.11257	-3.22513	-7.450
0.5	11.56285	6.281424	3.640712	-1.64071	-4.28142	-9.552
0.6	13.67542	7.337709	4.168855	-2.16885	-5.33770	-11.675
0.700000	15.78799	8.393994	4.696997	-2.69699	-6.39399	-13.78
0.800000	17.90056	9.450278	5.225139	-3.22513	-7.45027	-15.90
0.900000	[illegible]	10.50[illegible]	5.753[illegible]	-3.75[illegible]	-8.50[illegible]	[illegible]
1	22.1257	11.56285	6.281425	-4.28142	-9.56284	-20.1

LIMIT LOAD FACTOR CONDITIONS ALTITUDE 50,000 (FEET)
DUE TO MAX AND MIN CL'S

MACH #	CL = 1.5	CL = -1
0	0	0
0.1	1.383276	-0.92218
0.2	5.533105	-3.68873
0.3	12.44949	-8.[illegible]
0.4	22.13242	-14.7549
0.5	34.5819	-23.0546
0.6	49.79795	-33.1986
0.700000	67.78055	-45.1870
0.800000	88.52969	-59.0197
0.900000	112.0454	-74 6969
1	138.3277	-92.2184

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ALTITUDE 75,000 (FEET)

EFFECT OF WIND GUST

MACH #	VERTICAL MAGNITUDE OF GUST (FT/S) 40	20	10	-10	-20	-40
0	1	1	1	1	1	1
0.1	1.640621	1.320311	1.160155	0.839844	0.679689	0.359378
0.2	2.281242	1:640621	1.320311	0.679689	0.359378	-0.28124
0.3	2.921864	1.960932	1.480466	0.519534	0.039069	-0.92186
0.4	3.562485	2.281242	1.640621	0.359378	-0.28124	-1.56248
0.5	4.203106	2.601553	1.800777	0.199223	-0.60155	-2.20310
0.6	4.843727	2.921864	1.960932	0.039068	-0.92186	-2.84372
0.700000	5.484349	3.242175	2.121087	-0.12108	-1.24217	-3.48434
0.800000	6.12497	3.562485	2.281242	-0.28124	-1.56248	-4.12497
0.900000	6.765591	3.882796	2.441398	-0.44139	-1.88279	-4.7655
1	7.406213	4.203106	2.601553	-0.60155	-2.20310	-5.4062

ALTITUDE 75,000 (FEET)

LIMIT LOAD FACTOR CONDITIONS
DUE TO MAX AND MIN CL'S

MACH #	CL = 1.5	CL = -1
0	0	0
0.1	0.419468	-0.27964
0.2	1.677873	-1.11858
0.3	3.775216	-2.51681
0.4	6.711494	-4.47432
0.5	10.48671	-6.99113
0.6	15.10086	-10.0672
0.700000	20.55395	-13.7026
0.800000	26.84598	-17.8972
0.900000	33.97695	-22.6513
1	41.94684	-27.9645

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EFFECT OF WIND GUST ALTITUDE 100,000 (FEET)

VERTICAL MAGNITUDE OF GUST (FT/S)

MACH #	40	20	10	-10	-20	-40
0	1	1	1	1	1	1
0.1	1.193195	1.096597	1.048299	0.951701	0.903402	0.806805
0.2	1.386389	1.193195	1.096597	0.903402	0.806805	0.613610
0.3	1.579584	1.289792	1.144895	0.855104	0.710208	0.420419
0.4	1.772778	1.386389	1.193195	0.806805	0.613610	0.227221
0.5	1.965973	1.482987	1.241493	0.758506	0.517013	0.034026
0.6	2.159168	1.579584	1.289792	0.710208	0.420416	-0.15916
0.700000	2.352362	1.676181	1.338091	0.661909	0.323818	-0.35236
0.800000	2.545557	1.772779	1.386389	0.613610	0.227221	-0.54555
0.900000	2.738752	1.869376	1.434688	[illegible]	[illegible]	[illegible]
1	2.931946	1.965973	1.482987	0.517013	[illegible]	[illegible]

LIMIT LOAD FACTOR CONDITIONS ALTITUDE 100,000 (FEET)

DUE TO MAX AND MIN CL'S

MACH #	CL = 1.5	CL = -1
0	0	0
0.1	0.131096	-0.08739
0.2	0.524386	-0.34959
0.3	1.17987	-0.78658
0.4	2.097546	-1.39836
0.5	3.277416	-2.18494
0.6	4.719479	-3.14632
0.700000	6.423736	-4.28249
0.800000	8.390186	-5.59345
0.900000	10.61883	-7.07922
1	13.10967	-8.73977

Appendix A.10

<u>Airframe Cost</u>

The following equations, found in Reference 24, allow an estimate to be made for cost associated with the airframe only. The equations give an estimate in 1970 dollars, so the results had to be adjusted to 1990 dollars.

Engineering hours: $E= .0396*A^{.791}*S^{1.526}*Q^{.183}$

where A = airframe weight(lbs)

S = maximum speed at design altitude (knots)

Q = cumulative quantity produced

Development support cost: $D= .008325*A^{.873}*S^{1.89}*Q^{.346}$

Flight test operations cost: $F= .01244*A^{1.160}*S^{1.371}*Q^{1.281}$

Tooling hours: $T= 4.0127*A^{.764}*S^{.899}*Q^{.178}$

Labor cost: $L= 28.984*A^{.740}*S^{.543}*Q^{.524}$

Quality control: $Q/C= .13*L$

Material cost: $M= 25.672*A^{.689}*S^{.624}*Q^{.792}$

Solving the above equations for the aircraft,

assuming A=3464 lbs (Figure 4.8) and S=346 knots (Mach .44 at 100,000 ft), leads to the following.

E*($/hr)+D+F+L+Q/C+M = $10,398,187 (1970 dollars)

Converting this to 1990 dollars yields a total airframe cost of $40,237,700 (assuming a constant inflation rate of 7%).

It is important to realize that this figure is probably extremely conservative considering that this aircraft is not conventional in any sense. For example, the low airspeed (S) and low weight (A) make this aircraft appear, to the equations, to be the equivalent of a small military aircraft.